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HEAVY-LIFT TIP TURBOJET ROTOR SYSTEM
VOLUME III
DESIGN LAYOUT STUDIES

October 1965

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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VOLUME III

DESIGN LAYOUT STUDIES

Hiller Engineering Report No. 64-43

Prepared by

Hiller Aircraft Company, Inc.
Subsidiary of Fairchild Hiller Corporation
Palo Alto, California

for

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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SYMBOLS

c	Specific heat, BTU/lb-$^\circ$F.
D_p	Diameter, Primary Flow
D_{r_j}	Turbojet inlet momentum, lb/sec.
D_{r_s}	Secondary-flow inlet momentum, lb/sec.
D_s	Diameter, Secondary Flow
F_{ej}	Gross thrust, ejector, lb.
F_j	Gross thrust, turbojet, lb.
$F_{N_{ej}}$	Net thrust, ejector, lb.
F_{N_j}	Net thrust, turbojet, lb.
h	Film coefficient for thermal convection heat transfer, BTU/hr-ft^2-$^\circ$F.
Jc_p	Mechanical equivalent of heat, 778 ft-lb/BTU, constant pressure
K	Thermal conductivity, BTU/hr-ft^2 ($^\circ$F/ft)
P_o	Pressure, outside, lb/ft^2
P_p	Pressure, primary, lb/ft^2
P_s	Pressure, secondary, lb/ft^2
q	Rate of heat flow, BTU/hr.
T_s	Total temperature, $^\circ$R, secondary (cooling air)
T_p	Total temperature, $^\circ$R, primary (engine exhaust)
V_o	Velocity, outside, ft/sec.
V_j	Velocity, jet exhaust, ft/sec.
$\dot{w}$	Weight flow rate, lb/sec.
$\dot{w}_s$	Weight flow rate, secondary, lb/sec.
$\dot{w}_p$	Weight flow rate, primary, lb/sec.
ϵ	Emmisivity
η	Efficiency
γ	Weight density, lb/ft^3

1.0 SUMMARY

This volume describes the design layout studies performed by the Hiller Aircraft Company Design Department during the preliminary design of a tip turbojet rotor system applicable to a helicopter of 60,000 pounds to 80,000 pounds gross weight. The basic rotor system geometry for which these studies were performed was established by means of a parametric design study, as described in Reference 9.

Design investigations were directed primarily towards the components above the attachment of the rotor system to the airframe. These included rotor hub and blade retention configuration, rotor blade structural arrangement, power plant installation, flight controls, and airframe/rotor mounted fuel, lubrication, electrical, engine starting, and power management systems. Consideration was also given to tail rotor and accessories drive systems.

These design studies and associated stress and weight analyses have established the practicability of the design and fabrication of the tip turbojet rotor system, as well as providing verification for the systems' weights established by the parametric design study weight equations. Development of a tip turbojet helicopter of 60,000 pounds to 80,000 pounds gross weight has been found to be well within the state of the art of all technologies associated with the design and fabrication of an aircraft of this type.

2.0 CONCLUSIONS

2.1 Development of a tip turbojet helicopter of 60,000 to 80,000 pounds gross weight is concluded to be well within the state of the art of all technologies associated with the design and fabrication of an aircraft of this type.

2.2 A rotor system can be designed for this helicopter which provides freedom from ground resonance and flutter, and whose weight is approximately the allowable weight established in the parametric study.

2.3 The rotor blade design requirements are established by in-plane and flapwise stiffness requirements, and not by centrifugal restraint of the tip mass.

2.4 All components of the rotor system can be manufactured by conventional techniques.

2.5 Extensive use of titanium alloys in the rotor blades, hub, and mast would yield the required structure at minimum weight, and simultaneously provide excellent fatigue properties and corrosion resistance.

2.6 A gimbal-mounted rotor is the most advantageous for this application.

2.7 A hollow rotor mast can be designed which is dynamically compatible with the rest of the rotor system, which transfers the applicable loads, and which permits the transfer of system fluids and electrical power from the fuselage to the rotor blades.

2.8 Primary and secondary power sources for tail rotor, electrical, hydraulic, and accessory requirements can be achieved for minimum weight by the use of an auxiliary power unit and a small, mast-driven gearbox.

2.9 Conventional flight controls, using hydraulic boost cylinders, can be employed for this heavy lift tip powered rotor system.

2.10 It is practical and preferable to store all fluids and to originate all electrical power in the fuselage, and to transfer these to the rotor mast and rotor blades by a system of rotating manifolds, swivels, and slip rings.

2.11 An over-under engine configuration in preference to a side-by-side configuration results in lower engine installation weight and reduced complexity.

2.12. The mounting of turbojet engines and nacelles at the rotor tips and the design of all engine systems relating to the installation are straightforward and pose no insurmountable problems.

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2.13 The power management system study concludes that the system should be optionally manual or automatic and should provide fail-safe and back-up features for all its functions.

2.14 All aspects of the electrical system, including rotor tip located components, are within the present-day state of the art.

3.0 DESIGN STUDIES

The establishment of the optimum tip turbojet rotor system applicable to a helicopter of 60,000 to 80,000 pounds gross weight entails the careful blending of aerodynamic, structural, and mechanical characteristics. Hence, design layout studies were initiated simultaneously with parametric and dynamics analyses to assure adequate consideration of all important factors.

The preliminary design of a helicopter rotor system is concerned with the establishment of the basic rotor geometry including rotor diameter, number of blades, chord length, blade structural arrangement, blade to hub retention, rotor hub to shaft mounting, and method of obtaining rotor control.

In addition, in the case of a tip turbojet rotor, consideration must be given to the mounting of power plants at the rotor blade tips, with special attention to environmental conditions such as cycling and fluctuating gyroscopic moments, centrifugal loads, air flows, and the transfer of power plant systems from the fuselage to the blade tips.

The design and layout studies program was begun with the basic philosophy that all rotating components (the rotor, rotor mast structure, power plant systems' components, and control components) would be designed as flight hardware. This assures that the critical components will receive extensive whirl testing prior to the flight testing. The resultant minimum redesign for flight provides very significant savings in overall program cost and development time.

A program of periodic design reviews was conducted during the study to assure that the basic design criteria of functionability and reliability be met and that maintainability, manufacturability, low cost, and low weight be most effectively balanced. These reviews served also to assure coordination of the various design interfaces.

A preliminary aircraft configuration study was made to evaluate the relationship of the complete rotor system to the aircraft. This was necessary to determine the significance of rotor mast height, to establish optimum positioning of the various systems, and to provide a rotor system that will be compatible with the ultimate flight vehicle.

Three primary areas were studied: namely, rotor systems, power plant systems, and electrical systems. These extensive design studies treat all the important factors involved in the successful application of tip turbojet propulsion to helicopters. Design and optimization of the various systems were conducted, as described in the succeeding paragraphs of this report.

3.1 Rotor System

The determination of the optimum rotor system required careful consideration of all currently successful rotor systems, in terms of their applicability to the unique conditions and requirements of a rotor-tip-powered helicopter. The resultant configurations of hub, rotor blade, and rotor mast so derived constitute an optimization with respect to the characteristics of functional performance, reliability, weight, cost, and maintainability. The rotor system assembly is shown in Figure 1.

3.1.1 Hub Assembly

The several types of rotor hubs considered, and their applicability to this installation, were:

a) <u>Rigid Rotor</u>. Theoretically, the rigid rotor hub is the simplest mechanical arrangement; however, it is not considered a thoroughly proven system since investigations as to its structural integrity in extreme maneuvers is incomplete (Reference 19). Further, its theoretical simplicity is complicated by the necessity for the addition of an isolation system to preclude mast motion effects on control inputs.

b) <u>Articulated Rotor</u>. Full or partially articulated rotor arrangements were eliminated from consideration on several bases:

   1) The inherent low flapwise stiffness in the fully articulated rotor required heavy droop stops.

   2) In the event of an engine failure, the reduced lag restraint of the fully articulated system would cause a shift of the center of gravity of the rotating system, resulting in a vibratory condition of unknown magnitude.

   3) Undesirable ground resonance characteristics.

   4) Relatively complicated hinged retention systems are involved.

c) <u>Teetering Rotor</u>. This type is applicable only to a two-blade hub; and since the stability and control analysis had indicated that a minimum of three blades was required, this type of hub was eliminated.

d) <u>Gimbal or Universally Mounted Rotor</u>. This type of hub incorporates some details which tend to diminish the number of unknowns and is backed by a substantial amount of supporting data. The addition of an elastic restraint in combination with the gimbal mounting of the hub produces an arrangement that combines most of the features desired for satisfactory control and low vibration and stress levels, at an acceptable weight.

On the basis of the foregoing considerations, and in the interest of economy associated with reduced developmental risk, it was decided to concentrate design effort in the gimbal-mounted rotor. A constant velocity universal joint was also studied for this application, but because of its inherent complexity and greater weight was discarded in favor of the simple, conventional gimbal ring.

The preliminary design of the rotor hub is presented in Figure 2. The hub gimbal point is vertically located at the blade centers of gravity, 7.5 inches above the axis of blade intersections. This location should provide minimum vibratory loads, but it presents some construction complications, since it lies in the same plane as the upper hub retention plate. In addition, removal of gimbal bearings for maintenance requires a single sequence of disassembly. During detail design of the rotor, consideration will be given to locating the gimbal point closer to the blade intersection axis for ease of assembly and maintenance.

Tests on Hiller's small UH-12L series helicopter indicated negligible change of vibratory levels when the teetering point was displaced eccentrically a small distance from the blade mass center of gravity. The gimbal ring shown in Figure 2 is a relatively simple forging. However, for the aircraft design phase, the twin bearing lugs will be placed on the gimbal ring and the single mating lug will be on the mast and/or hub. This will simplify the mast forging and the rotor hub ring.

3.1.2 Rotor Blade Assembly

3.1.2.1 Blade Retention

The high centrifugal forces generated by the large rotor blades and the tip-mounted engine installations result in a requirement for very large thrust bearings in the retention system and cause control loads of high magnitude. However, these problems are alleviated by the use of a system involving a tension-torsion link and radially loaded bearings. Several blade retention concepts were studied in order to select the most advantageous system.

The layout studies indicated that the different types of hub and root retention systems analyzed presented some difficulties which could be traced to the size and mode of fixity required. The in-plane stiffness required was found to be several times higher than the flapwise stiffness, indicating that for weight minimization the distribution of material should not be uniform about the feathering axis.

The types of retention arrangements studied and the significant characteristics of each are as follows:

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a) Internal Strap (Figure 3)

This configuration consists of a cylindrical section hub with four hollow, cylindrical spindles radiating from the center. Two span-wise-located roller-type pitch bearings roll on the spindle and are mounted in a larger cylindrical trunnion attached to the rotor blade. A tension-torsion strap package of either flat sheet metal laminations or a continuous wrap of urethane-coated wire is attached to both the hub and the blade on the centerline of the spindle. This strap arrangement transfers the centrifugal force into the rotor hub, and the roller bearings take only radial loads.

b) Modified Internal Strap (Figure 3)

A modified version of the previous configuration uses the trunnion as a transition section between the blade airfoil and the cylindrical shape, over a larger inboard pitch bearing. This is done to avoid a drop of the stiffness curve section properties between the hub spindle and the trunnion inboard end. The spindle is of sufficient size to meet the stiffness requirement, and the large pitch bearing in the large trunnion end increases the stiffness above the required amount.

c) External Strap (Figure 3)

The external strap retention method employs the same spindle-trunnion combination as the internal strap arrangement except that the two tension-torsion strap packages are located externally in the chord plane, one on each side of the trunnion.

d) Laminated Bearings (Figure 4)

In this configuration, the hub consists of a hollow cylindrical section with integral hollow cylindrical spindles radiating from the center. A complete departure from conventional rotor retention was investigated with radial and thrust bearings made from laminations of thin metal and thin rubber sheets bonded together. These replace the roller bearings and the tension-torsion strap packages and would not require lubrication.

e) Flexural Hinge (Figure 4)

This arrangement is another departure from any conventional configuration. It consists of four diagonally opposed tension-torsion strap packages attached to a cylindrical hub. These straps angle outboard toward an apex centered in the blade retention spindle. The spindle is held in the hub by a spherical bearing and is located midway between the straps at the hub end. The hub consists of a hollow cylindrical section with the strap attachment lugs machined as integral parts. This hub is longer and heavier than the other blade retention systems due to the widely displaced strap attachments.

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f) <u>Stub Blade Retention System</u> (Figure 4)

The studies of the foregoing configurations led to the conception
of a retention system which provides high chordwise stiffness
while maintaining adequate flapwise stiffness at minimum weight.
This system, called the Stub Blade Retention System, is superior
to the other systems considered from the standpoints of weight,
manufacturing feasibility, cost, flexibility of design, growth
potention, and relative aerodynamic cleanliness. It is described
in the following paragraphs.

The centrifugal forces acting upon a blade are reacted by a tension-
torsion strap located concentric with the feathering axis. Since it is
desired to keep the feathering moments and the mass balance to a minimum,
the location of the feathering axis is placed as near as possible to the
blade aerodynamic center axis. In this case, it is at the 25-percent-
chord location.

The installation of a tension-torsion strap concentric with the feather-
ing axis governs the size of the bore of the radially loaded bearings,
which are sized to provide clearance for operation and removal of the
strap. In order to provide space for the installation of the bearings,
the bearing supports, and the tension-torsion element, a cutout was in-
corporated at the inboard front end of the blade, the remaining struc-
ture offering sufficient geometry for an efficient beef-up to bring the
chordwise and flapwise section properties to the desired values.

The attachment of the rotor blade to the hub is made through a complemen-
tary structural element which is a component of the hub system, and which
is named a stub blade because of its general shape. For purposes of its
analysis, the stub blade and the inboard extreme of the rotor blade are
treated as components of the blade retention system (see Figure 5).

Once the general concept was established, the dynamic studies produced
data that allowed the proportioning of the retention elements to give
minimum weight. Some critical considerations, such as the stiffness
or spring rate of the bearings and bearing supports, will need experi-
mental substantiation before optimization of the retention system geo-
metry.

3.1.2.2 <u>Blade Design</u>

The design of very large rotor blades involves problems that require
new approaches to their solutions. Preliminary investigations showed
that large size effects preclude the arbitrary use of a scaled-up ver-
sion of an existing blade without a prohibitive weight penalty. Good

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design required that the major parameters of the blade system be so established as to provide a maximum of flexibility in the design of specific blade components. From the standpoint of producibility, the manufacture of rotor blades of large size presented particular problems due to the limitations and capacity of existing fabricating equipment. This necessitated that the detail design of parts be such that the final configuration of the whole assembly would result in a minimum of practical compromises. For example, by properly selecting the fabrication techniques a reduced number of splices could be obtained, thus decreasing the number of fatigue problem areas. This is an important factor when the magnitude of loads and stiffness is considered, since the local beef-ups at the splices can add considerable weight. The foregoing are among the factors which must be considered when idealized structures are used as a basis for comparison, in order to make decisions based on feasible configurations.

3.1.2.2.1 Blade Structure Design

The classical works of structural weight estimation and optimization (References 6, 13, 14, and 17) analyze some theoretical and empirical approximations to the laws governing weight as a function of strength. While the structural stability factor can be considered in a substantially rational manner, the complexity of the flutter factors prevents anything but a rough approximation. When such studies are modified in an attempt to incorporate other variables, such as specific vibratory response modes, the problem ceases to be manageable in regard to the derivation of an analytical tool for design approach purposes.

In view of the large number of variables and the lack of knowledge of the relative importance of these on the weight picture, a process of iteration, starting from simplified assumptions, was applied in this program in order to define the preliminary requirements of the most significant blade parameters. For the first exploratory study in the iteration process, values were based on the proposed blade geometry and dimensions previously considered in Reference 16. These were subjected to an investigation to determine the feasibility of the proposed configuration. Because the mass of the engine installation could be lumped into a concentrated unit and its weight was known with some degree of accuracy, the rotor spar caps, webs, and skins were the only components varied in the iterative process. Assuming different blade tip weight to blade weight ratios for a given tip speed, it was possible to find some stiffness parameters which defined the blade section properties (Reference 9).

It was readily apparent that a constant blade chord would simplify the installation of the tip mounted engines by allowing the use of the maximum available chord, and corresponding airfoil depth. This results in

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lower weight of engine mount structure and the associated retention structure in the blade, which in turn results in a lower required weight of rotor blade structure. The comparatively small increase in aerodynamic efficiency of a tapered blade is not considered sufficient to offset these weight advantages.

Since the stiffness required is a function of the mass and its distribution, a first analysis was made taking the blade tip as a starting point. The net cross-sectional area required to resist the centrifugal loads was computed for several blade stations. Stiffness checks were made, and, after making allowances for practical thicknesses, a rough mass distribution plot was made. Natural frequencies were checked and some cross sections were modified. This process was repeated until further changes did not show significant weight savings. At each stage, the location of the center of gravity of the blade section was determined to insure its falling as close to the 25-percent-chord point (blade feathering axis) as possible. Alternate materials, 301 stainless steel and Ti-8Al-1Mo-1V titanium alloy, were considered.

At the time these studies were made, the flutter analysis results were not known, and the bases for torsional stiffness criteria were the ones described in Reference 2. The skin thicknesses used were consequently the ones that assured flutter-free operation throughout the complete feathering range spectrum. The degree of conservatism is not well known, but indications are that this choice of relatively thick skins might lead to a relaxation of the final location of the blade section center of gravity from the standpoint of flutter considerations, thus leaving the final chordwise mass unaltered.

A basic NACA 0015 airfoil was used for all section properties analyzed. Since the maximum section thickness has the most profound effect on flapwise frequencies, it is felt that moderate airfoil shape changes can be made without affecting the weight figures to any important degree, as long as the airfoil thickness is maintained at a value of 15 percent of the chord length. In considering the design of the blade leading edge, any sharpening of the leading edge would increase the difficulty of housing the plumbing of the engine services, which would force the nose spar to be shifted aft, resulting in an undesirable weight penalty.

For all configurations studied, the leading edge was assumed to be structurally inactive. This was done because of the simplification involved in routing fuel, oil, and electrical systems through the airfoil nose. The need for easy access to these systems precludes the use of full-span structural panels without undue complications. This consideration, in addition to the consideration of ease of damage repair, favors the use of short segments of leading edge skins. The leading edge covers, because of their thickness, will provide a good support for any doubler material to be used as mass balance counterweights, if needed. Any required

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adjustment can be made with ease by removing two lines of fasteners holding the cover plates to the main structure and adding or removing doublers to the inside surface.

The configurations studied included two types of trailing edge structure. One type included an aft structural box extending from root to tip, connected with the main forward box beam. The other type was a combination of a forward box spar with a removable segmented trailing edge made up of rectangular segments interrupted in the spanwise direction.

For the purpose of this study, both blade designs were considered extensively. One type is identified as the "fully effective blade," its structure extending from the 5-percent-to 100-percent-chord location. The other type is identified as the "partially effective blade," having an active structure extending from the 5-percent- to 50-percent-chord location. These two types are worthy of complete discussion, in view of their effects upon the blade structural configuration and weight, and are treated in the paragraphs which follow.

Figures 6 and 7 show weight comparisons for the fully effective and partially effective blade designs as a function of chord length for two values of rotor radius and stiffness and for both steel and titanium. These figures show that for either type of blade, the weight decreases as chord increases in the range of five- to eight-foot-chord lengths. This holds true for either steel or titanium, and for either a 56-foot rotor radius or a 65-foot rotor radius. Selection of the fully effective blade was made on the following bases:

a) Its weight is approximately equal to that of the partially effective blade.

b) The partially effective blade weight would grow due to local beef-up required at the root and tip.

The basic structure of the partially effective blade can be described as a two-spar box beam. A forward "I" section spar with a massive web is located at 5 percent of the chord; a second "I" section spar is located at 50 percent of the chord; a top and a bottom skin connecting forward and aft spar caps complete the structural box beam cross section. Some suitably located light frames transfer the air loads to the spars and also stabilize the skin and hold the blade contour. Two closing ribs, one at the inboard end and one at the tip, complete the basic blade box beam.

Study of this design approach indicated that the front spar, being relatively shallow, required quite a massive web to help build up the required cross-sectional area.

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In the fully effective blade design, as shown in Figures 5, 8, and 9, the primary structure is based on the two-cell-box, two-spar-beam concept. A third cell at the leading edge is made nonstructural, as noted previously, and completes the airfoil cross section. The two structural cells have a common web at the 50-percent-chord location, which was selected by the previously noted iterative optimization study involving stiffness, strength, and load path analysis. The location of the front spar was established based upon the need for ready access to electrical, fuel, and oil lines, which were located at the leading edge to alleviate blade balancing requirements.

The front structural cell, extending from the 5-percent-chord point to the 50-percent-chord point, constitutes the primary structure of the rotor blade, to which the root retention and engine mount loads are applied. It consists of a channel-shape member including upper and lower skins and aft spar web, formed as a unit and attached to the front spar to form the perimeter of the cell. The wall thickness of this form changes by gradual taper from the blade root to the tip. This can be accomplished without the use of transverse joints if the member is made by the roll-shear method, or with some joints if made by welding, bonding, or a combination of the two. Tapering in the jointed version would be accomplished by chem-milling, leaving buildups where the joints are made. Local doublers at the root and tip are added by bonding and mechanical fasteners if needed. Attachment of the beef-up material by ultrasonic welding or by electron beam welding out-of-vacuum are possibilities that have not been explored thoroughly and will deserve further consideration.

The trailing edge cell constitutes the second component of the primary structure. Its cross section appears as a hollow wedge formed by top and bottom honeycomb skin panels. The cover system is composed of flat skin panels made by bonding a titanium outer facing and an aluminum alloy inner facing with an aluminum nonpermeated honeycomb core which extends to the edges of the facing. The edges can be filled with resin or covered to prevent exposure of the core to the environment. This wedge-shaped box is attached to the forward cell by extensions of its honeycomb skin panels which overlap a joggled area of the front cell skins. The main structural function of the aft cell is its contribution to the in-plane and torsional stiffness of the blade. A concentration of cap material at the trailing edge effectively adds to the chordwise properties, and also serves as a cover plate for closure of the aft edge of the cell.

The structure formed by the front and aft cells is closed at the ends by bulkheads; another bulkhead is located at the change of section in the blade retention area. A series of ribs at both the front and aft cells transfers the air loads, maintains the contour of the airfoil, and stabilizes the skin. Skin beef-up, described previously, together with added material at the spar caps at the blade ends, provides a load diffusion path for the blade retention and engine pickup loads.

Design of the blade root structure, Figure 5, provides for omission of a portion of the forward section of the root and the addition of a secondary spar to accommodate the blade retention system. The secondary spar closes the face of the remaining structure. This additional spar, together with some trailing edge skin beef-up and doublers, provides the required strength and stiffness. The inboard end of the blade contains a hinge bearing support fitting which also incorporates a control rod attachment fitting. This assembly is attached to a trailing edge closing rib. The second bearing fitting of the blade retention system is attached to the intermediate closing rib installed at the section where the blade extends in full to the leading edge. Axial restraint of the blade is accomplished by a pin which ties a tension-torsion strap element from the hub to the blade through fittings attached to the blade skin and spars.

Accommodations for engine retention are provided at the rotor tip, as shown in Figure 8. One set of fittings is attached to an extension of the blade skin and doublers and distributes the main engine loads from the forward engine mount to the spars. The aft engine mount support fitting, likewise attached to the skin and doublers, is immediately adjacent to the spar which is located at the 50-percent-chord point.

3.1.2.3 Manufacturing Considerations

During the process of optimization of the blade structure, manufacturing and operational considerations determined some limits. Fabrication based on shapes extruded from steel and titanium alloys was considered to be a feasible means of manufacture, although some producers cannot handle sizes as large as required; even so, splicing by welding and/or bonding is an entirely acceptable means of creating parts of the required proportions.

In order to minimize fatigue problems, the process of bonding parts together is a cardinal part of the design philosophy concerning joints. While there is abundant experience on bonding moderately sized parts, certain contoured and heavy sections must be investigated to assure that their bonding will meet strength requirements. Solutions involving a combination of mechanical fasteners and bonding with definable fail-safe margins is also a possibility that has been contemplated. Among the joining techniques to which some attention has been directed, electron beam welding (both inside and outside a vacuum chamber), and ultrasonic welding are likely to be applicable. It is considered that for some techniques, if the advantages warrant, justification by testing will be sought.

The shape of the main structural cell of the blade needs careful study because of its large size. Consideration has been given to the process

of working a billet into a tapered wall tubular form by using a technique
of shaping a rotating workpiece by a combination of rolling and shearing
action. A secondary operation would modify the circular cross section
into the required airfoil contour, including some details such as
longitudinal joggles and steps, necessary to provide flush overlapping
joints with the remaining components of the structure. Depending on
the final size of the basic structural box beam, either a single tube
or a side-by-side accumulation of properly shaped elements may be used.
The desired amount of twisting of the resulting tubular shape will be
performed before or during subassembly in the master jig, depending
on whether the components are closed or open shapes.

The blade structure presents areas where some subassembly operations
include bonding. It is not desirable to devise an assembly sequence
with more than two or three additional curing cycles, since to do so
would degrade the strength of the previously bonded subassemblies.
Careful planning in the design of details will assure the desired high
degree of structural efficiency. The use of the autoclave curing chamber
and/or localized heat and pressure systems are approaches which will be
investigated.

The structural concept shown in Figures 5, 8, and 9, is based upon a
tubular shaped element whose front face has been removed to allow
assembling the internal details. This also allows the attaching of
the front spar to the doublers and skin by simple overlapping.

3.1.2.4 Choice of Basic Structural Material

After the process of blade optimization was completed, enough data and
results had been accumulated to permit drawing some important conclusions
regarding materials of construction. The configurations on which the
study was based are felt to reflect representative types of designs
resulting from the process of synthesis starting from specifications
and feasibility considerations. Material selection played an important
part in this study, with both steel and titanium in the alloyed forms
being considered to the fullest extent. The use of aluminum was restricted
to honeycomb core applications and internal facings of honeycomb panels.
The reason for this is the low fatigue strength-to-weight ratio of
aluminum in comparison with high strength steel and titanium. An equa-
tion derived in the performance parametric study (Reference 9) indi-
cates that a pound of weight added at a station in the area of the
engine installation center line, would require an additional 1.2 pounds
of blade structural support, making a total increment of 2.2 pounds
for the rotor blade basic weight. While this structural weight penalty
will reduce in magnitude when considering sections closer to the rotor
center of rotation than the rotor tip, it emphasizes the value of weight
saving in the rotor blade.

Comparison of structural materials indicated which ones would result in a minimum weight structure (Figures 10 and 11). The selection is based primarily on stiffness and fatigue properties. Half-hard 301 stainless steel and Ti-8Al-1-Mo-1V duplex annealed titanium alloy properties were considered in the design studies. While the cost of titanium is higher than steel, it was felt that, on the basis of weight saved and fatigue life advantage, the titanium alloy could prove to be the ideal material for the rotor blade.

There is a wide acceptance of titanium as major component material for high performance aircraft. The state of the art at present is such that the manufacturing of titanium parts requires routine-type procedures which are well known and proven. Many research and development studies and some projects (including the North American B-70 supersonic transport and F-100, Lockheed A-11, McDonnell Phantom II, Northrop F5B, Northrop Boundary Layer Control aircraft, Boeing, Convair and Douglas jet transports, and several engines and missiles) give support to the use of titanium as a reliable means of saving weight (Reference 20). In some instances, direct substitution of titanium alloy for high-strength steel was possible, with consequent weight saving.

3.1.3 Rotor Mast Assembly

In addition to its conventional function, the rotor mast of this helicopter is required to act as a transfer device for electrical and engine fluid systems, and to drive an accessory gearbox. The mast assembly is shown in Figure 1.

3.1.3.1 Mast Tube

This tube was designed with several basic functional considerations:

a) It must allow all the power plant servicing lines to be carried internally.

b) It must carry all the lifting and bending loads of the rotor.

c) The tube must be of sufficient stiffness so that it will not resonate with the major frequencies of the main rotor.

d) It must have sufficient torque capacity to drive a gearbox for basic flight accessories.

e) It must adequately transfer blade control loads and carry the rotor spring restraint loads.

In addition to the above requirements, minimum weight, reasonable ease of fabrication, ease of assembly or disassembly, and reasonable cost were given strong consideration.

Initially, a mast diameter was selected to provide sufficient room for passage of engine starting and servicing lines. Then, an arbitrary minimum wall thickness was selected to give a diameter to thickness ratio of about 60. The bending modulus of rupture for this ratio is approximately the same as the ultimate tensile strength. This tube was then checked for the maximum lifting and bending loads equivalent to a 2.5g flight load factor and a ten-degree tilt of the rotor plane. This analysis was followed by a check of the natural circular frequency of this shaft, which is dependent on the tube section properties, tube material, support point locations, and the location and weight of the rotor. The natural circular frequency was foun: satisfactory. The tube was then evaluated for torque, control element, and rotor restraint loads. The design evolved into an upset forging, titanium alloy tube of 15-inch outside diameter and 0.25-inch wall thickness.

3.1.3.2 Rotor Mast Mounts

The rotor mast is supported by two sets of bearings, as shown in Figure 1. Two supporting systems are considered here: one is the rigid system for the whirl stand, which has been shown by a dynamic analysis to be satisfactory; the other is the snubbing or elastomeric system for the flying aircraft. Basically, the upper bearings are designed to react radial loads, or loads normal to the mast. The lower set of bearings react the lifting thrust loads, downward rotor weight loads, and the radial loads. The upper bearing is a double row tapered roller bearing designed for an elastomeric strut system. A self-aligning bearing will replace this bearing in the whirl stand rigid system. The lower bearings are gimbal-mounted, in both systems; however, the gimbal is not required in the rigid system.

3.1.3.3 Tail Rotor and Accessory Drive

In conventional helicopter practice, the tail rotor and vital accessories are driven directly by the main rotor so that their continued power supply during autorotation is assured. Accessories of a less critical nature which are required only during normal flight are driven by the engine.

In the tip turbojet-powered helicopter, the tail rotor and vital accessories can be powered by a gearing system driven from the main rotor mast, so that the autorotational requirements can be met. However, the nominal speed requirements of the accessories is 6,000 r.p.m.; the speed requirement of the tail rotor drive shaft is 2,700 r.p.m.; and the speed of the main rotor shaft is 110 r.p.m. The large difference in speed between the rotor mast and the driven components results in a large torque multiplication at the rotor mast to drive the components. This establishes a trend toward a large, heavy gearbox; hence, it was desirable in this program to investigate minimizing the power taken from the main rotor and thus minimize this gearbox size.

Several arrangements were considered involving the use of an auxiliary power unit in conjunction with a mast-driven gearbox. The availability of an appropriate auxiliary power unit (APU) was established. Comparison was made on the basis of weight, system efficiency, and gearbox complexity. The criterion was that the aircraft be capable of safe autorotation under any single condition of hydraulic system, fuel system, or auxiliary power unit failure. In addition, the requirement was established that the pilot have the option of completing his mission with reduced directional control.

The arrangements considered were:

a) Tail rotor and all accessories driven by gearbox from the main rotor mast for all flight conditions. No auxiliary power unit.

b) Tail rotor and minimum accessories to support autorotation driven by gearbox from the main rotor mast for all flight conditions. Primary accessories driven by auxiliary power unit.

c) Tail rotor driven by gearbox from the main rotor mast only during autorotation. Minimum accessories to support autorotation driven by gearbox from the main rotor mast for all flight conditions. Primary tail rotor and accessory drive by auxiliary power unit.

Case a) results in the heaviest drive system, since all accessories and the tail rotor are driven by the mast, requiring a large and heavy gearbox. Case b) is also a heavy system in that the APU system weight is added, although the main gearbox weight is only moderately reduced. Case c) results in the lightest and most efficient arrangement. In this case, the tail rotor and the major portion of the accessories are driven by a single APU. The weight of the APU is offset by the mast gearbox weight saving, since in case c), the gearbox needs supply only about one-third as much tail rotor and accessory power; hence, the gearbox size is greatly reduced relative to case a). Although this study was not rigorously pursued in detail, the weight trends were clear. Further, the usefulness of the auxiliary power unit to provide engine starting air and ground power were considered as strong favorable factors in its behalf.

The main rotor mast accessory gearbox is fuselage-mounted and driven from the bottom of the rotor mast through a flexible coupling. A tail rotor drive shaft off the accessory gearbox connects with the APU-driven tail rotor drive shaft through an overrunning clutch and will provide tail rotor power in case of APU failure.

3.1.3.4 Transfer System

As shown herein, the source of all engine fluid systems and of the aircraft's electrical system is in the fuselage. The fluids and electrical energy are then transferred from the stationary fuselage into the

rotating rotor mast by the transfer system. Inside the mast, the systems
are led to the top, then out the rotor blades to the engines. The trans-
fer system assembly is shown in Figure 12.

The basic design concepts involved in the transfer of the various systems
from stationary supply sources into the rotating rotor mast dictate that
this transfer be accomplished with a minimum of pressure loss or leakage
or, in the case of the electrical system, minimum insulation or arcing.
For all systems, the efficiency and quality of the transfer are enhanced
if the mast surface speed at the point of transfer is kept as low as
possible. For this reason, the mast diameter is reduced in steps near
the bottom of the mast, and each system transfer occurs at a different
step.

Since the oil replenishment system involves a single, very small line,
it is transferred into the bottom center of the mast. The electrical
slip ring assembly is assigned the preferred transfer location, the mini-
mum diameter step at the bottom of the mast. Next above is the fuel
transfer step, and then the starting air transfer step. Appropriate
isolation will be provided between the fuel transfer system and the slip
ring assembly.

Lines from the various chambers at the bottom go up through the mast
into the distribution manifolds attached to the top of the shaft and out
into the blades. At the top of the mast there is no sealing problem,
since the rotor turns with the mast. The distribution chambers at the
top of the rotor mast are stacked together in the same manner as at the
bottom of the mast. The first distribution chamber above the top of
the mast proper is the engine starting air distribution chamber. This
is followed by the electrical wiring which emerges between the air
chamber and the first fuel distribution manifold.

Fuel gate valves are attached to the fuel manifold chambers between the
manifold and the fuel lines to the blades. A second fuel distribution
manifold is mounted above the first distribution manifold with another
fuel gate valve between the two manifolds. The two manifolds and sep-
arate fuel supply lines provide dual fuel system reliability, as shown
in the fuel system description. The remaining line is the oil replen-
ishing system line which goes up through the middle of all the manifolds
into its distribution manifold at the top. Consideration was given to
saving space and weight by combining the rotary joints and the distri-
butors and mounting them on the upper part of the mast below the con-
trols; however, this arrangement was not satisfactory from a service and
maintenance viewpoint, since rotor mast removal would be required for
maintenance.

3.1.4 Flight Controls

A conventional, hydraulically operated flight control system has been

selected, Figures 13 and 14. This consists of the usual swashplate actuated by three dual hydraulic boost cylinders. Two of the dual cylinders will actuate longitudinal cyclic control, one dual cylinder will actuate lateral cyclic control, and all three cylinders will actuate collective control. A dual hydraulic system is provided for reliability. The upper push rods from the swashplate actuate the rotor blades indirectly through walking beams and links. An intermediate stage of collar-supported isolation arms was necessary to prevent the tilting hub motions from feeding back and affecting the original input rod motions.

Brief studies were made to investigate placing the control rods within the mast, but the maintenance aspects of controls and the exterior mounting of fuel, air and oil transfer systems, with consequent greater sealing problems, ruled out this possibility. The aircraft controls from the cockpit to the dual hydraulic cylinders are planned as conventional push-pull rod systems with possible built-in artificial feel springs and a stability augmentation system.

The stability and control studies, Reference 10, indicates that a spring restraint system is highly desirable to improve control response. This is incorporated at each blade root with a spring (compression or tension) type strut, Figure 1. The spring axis is arranged normal to the radial line from the flapping axis so that a minimum weight spring assembly is obtained.

3.1.5 Hydraulic System

Two separate 3,000-p.s.i. hydraulic systems have been designed to obtain the reliability needed for the flight control system. One system is powered from the main rotor, and the other system is driven from the auxiliary power unit. These systems will also have sufficient capacity to operate landing gear retraction, wheel braking, and cargo hoist systems if required.

The selection of a 3,000-p.s.i. operating pressure is designed to give a minimum weight system consistent with good reliability. The fluid to be utilized is MIL-H-5606. Protective shielding against fire is planned for the hydraulic pump and system which is powered by the auxiliary power unit. The component parts, pumps, reservoirs, filters, relief valves, and check valves will be selected from flight-proven items used on present commercial and military aircraft. Provisions are planned for external outlet for ground checkout. The basic schematic for this system is shown in Figure 14.

3.2 Power Plant Installation

3.2.1 Introduction

This section of the Design and Layout Studies describes the design evolution of the power plant systems and installation in order that they meet the requirements of the tip turbine-powered helicopter. The following were investigated:

a) Engine location with respect to rotor blade chord, angle of engine centerline with respect to rotor centerline and with respect to rotor plane, and orientation of multiple tip engines.

b) Engine mounting configuration.

c) Nacelle configuration, including induction and exhaust systems and firewalls.

d) Engine compartment cooling.

e) Engine control system.

f) Fuel system.

g) Lubrication system.

h) Engine starting system.

i) Fire safety.

The power plant design studies were begun with certain basic criteria in mind. Engineering logic dictated that any component location in the 235g centrifugal field at the rotor tip would have to be well justified. Tip located components, besides requiring special qualification testing, would require close weight and center-of-gravity tolerances. Further, Reference 9 indicates that 1.2 pounds of back-up structure is required in the rotor blade for every pound of additional weight located at the rotor tip. Accordingly, all storage for engine systems (fuel, oil, and starting air if required) would be located in the fuselage and ducted out through the mast, hub, and rotor blades to the engines. Therefore, a prime requirement in any system was that it be compatible with the concept of routing its output from the stationary fuselage to the rotating rotormast, through a rotating manifold, swivel, or slip ring.

All systems are routed from the rotor hub to the tip engines through the rotor blades, and are group-mounted to the leading edge spar. Access to the systems for maintenance is accomplished by removal of the leading

edge. Figure 15 shows the systems' retention used in the rotor blades, and Figure 12 shows the method of transfer of system fluids from the fuselage to the rotor mast and the rotor blades. Figures 16 and 17 show the engine instrumentation and panel arrangement.

From the beginning of the study, it was apparent that a power source would be required in the fuselage to drive accessories and the direction-a.-control tail rotor. The accessories drive section of this volume shows that the system which evolved was a functionally optimized combination of a main rotor mast-driven accessory gearbox and an auxiliary power unit. In this system, the main rotor mast drives a gearbox providing power for alternate flight accessories and for driving the tail rotor in the event of APU failure. The auxiliary power unit drives the tail rotor, provides the power supply for accessories, and serves the additional function of providing air for engine impingement starting.

A power plant and systems installation isometric drawing is shown in Figure 18.

3.2.2 Engine Installation

The engine is the Continental model 357-1, a modified model J69-T-29 engine, rated at 1,700 pounds maximum thrust. Externally, the modified engine differs from the J69-T-29 in that it has a larger diameter (24-1/2 inches), incorporates an exhaust nozzle, and is approximately eight inches shorter in length.

The engine installation depended upon the results of wind-tunnel tests, the flutter analysis, and the preliminary rotor design studies; therefore, conclusions could not be reached until these analyses were complete. However, to expedite the program, a detailed engine installation study was begun based on the preliminary design philosophy expressed in Reference 16.

As a preliminary assumption, the engine center of gravity was placed at the 22-percent-chord location (the feathering axis of the rotor blades is at the 25-percent-chord location) to minimize flutter tendencies. The engine centerline was placed parallel to the main rotor blade chord line. Early analyses showed that the rated maximum engine thrust of 1,700 pounds precluded satisfying the helicopter requirements with a single engine per blade; hence all power plant installation investigations were conducted on two-engines-per-blade configurations.

For purposes of studying whether an over-under or a side-by-side engine arrangement would result in the best installation, a nacelle diameter of 30 inches in combination with the 22-inch diameter J69-T-29 engine was used. This nacelle diameter had been selected for aerodynamic

reasons and was to be used for the wind-tunnel model. Preliminary layouts confirmed that the inboard-outboard arrangement would result in increased complexity of installation in the areas of nacelle and firewall design and fluid line routing. The trends were so apparent at the outset that it was considered unwarranted to probe more deeply into this investigation. It was evident that an inboard-outboard arrangement would require a larger diameter cowl for a given engine to accomodate the engine mount space requirements, thus resulting in increased drag. Dynamic considerations showed no preference for either configuration and established that the blade-twisting moment associated with one-engine-out operation in the over-under arrangement would have negligible effect because of the stiffness of the blade. Accordingly, it was concluded that the over-under engine arrangement was the superior configuration, and design studies and layouts of the various power plant systems was begun on the basis of that arrangement. This conclusion was unaffected by the subsequent engine diameter change to 24-1/2 inches. The engine installation is shown in Figure 19.

3.2.3 Engine Mounting Configuration

The engine mount design involved continual close coordination with Continental Aviation and Engineering Corporation in the establishment of engine mount locations, to assure compatibility from the standpoint of both engine and helicopter requirements.

Initially, the engine mounting provisions consisted of two lugs located forward of the engine center of gravity and one lug aft. It was suggested to Continental that the main mounts be moved as near as possible to the engine center of gravity in order to put all the primary load into the main mounts. This would also place the main mounts in line with the rotor blade structure at approximately the feathering axis of the blade. The third mounting point would then act primarily as a stabilizing point and would take very low loads as compared to the main mount.

In the process of Continental's modification of the J69-T-29, the engine internal structure was modified to allow relocation of the engine shaft bearings. This modification required a very stiff section at the radial compressor which was located approximately at the engine center of gravity. Hence, it was found to be advantageous to locate the engine main mount lugs on this heavy engine frame. The placement of two lugs at this location on the inboard side of the engine, separated by a 60-degree angle, permitted the same engine mounting lugs to be used for both the upper and the lower engines. The 60-degree spread was determined to be large enough to accomodate the fore and aft couple resulting from gyroscopic forces as well as pitching loads. A larger spread would have increased the bending moment on the engine mount structure and would have been undesirable. A third, stabilizing lug was provided at the aft flange of the engine. A position at the forward end on the engine would have been

preferable from a temperature standpoint, since the aft flange is at or near a 900-degree Fahrenheit surface; however, a mount at the forward location would have interfered with the fuel control and with the routing of the engine service systems. Also, this location would not provide a good structural load path into the rotor blade.

A study was made of the directions of anticipated gyroscopic couples resulting from blade feathering, for either direction of engine rotation and for either direction of main rotor rotation. The mount studies and design have been based upon a clockwise engine rotation viewed from the aft end and counterclockwise rotor rotation viewed from above. Although the present engine direction of rotation is counterclockwise when viewed from the aft end, the mount solution reported herein is applicable for either direction of rotation.

After locating the engine and its mount points, a study was made of the mount structure requirements. The selected point for primary engine restraint was at the two forward lugs. At one of these lugs the engine is mounted solidly, providing for restraint of engine axial loads, centrifugal forces, gyroscopic couple loads due to pitching and flapping, and vertical loads. At the other forward lug, a connecting link is provided from the mount to the engine so that axial and centrifugal force and gyroscopic couple loads are restrained, but pure vertical loads are not, thus permitting the engine to expand radially. At the aft mount lug, a link is provided between the mount structure and the engine so that only centrifugal force loads are restrained, which allows the engine to expand axially. Engine mount loading was investigated for the following conditions:

a) Loads due to basic engine weights and turbine polar gyroscopic forces, with two engines operating.

b) Loads due to overspeed centrifugal force loads, with two engines operating.

c) Loads due to normal blade centrifugal force and gyroscopic forces, without cyclic control inputs, with two engines operating.

d) Loads due to normal blade centrifugal force and gyroscopic forces, without cyclic control inputs, with lower engine not operating.

e) Loads due to normal blade centrifugal force and gyroscopic forces, without cyclic control inputs, with upper engine not operating.

f) Loads due to normal blade centrifugal force and gyroscopic forces, with full cyclic control input, with two engines operating.

g) Loads due to normal blade centrifugal force and gyroscopic forces, with full cyclic control input, with lower engine not operating.

h) Loads due to normal blade centrifugal force and gyroscopic forces, with full cyclic control input, with upper engine not operating.

This investigation established the strength criteria to which the engine mount system was designed, resulting in the system described in Figures 20 and 21. The design is based upon the use of 6Al-4V titanium alloy, which results in a relatively lightweight engine mount capable of fitting within the aerodynamically determined cowl lines.

3.2.4 Nacelle Design

With the engine position established and the nacelle external lines determined to a large extent by aerodynamic considerations and the need to adhere as closely as possible to the configuration of the wind-tunnel model, the nacelle design study was focused mainly on fore and aft position, structural design, and compatibility with firewall, accessory, and engine services space requirements.

A preliminary nacelle layout was made to study the problems involved in designing the cowl structure, housing the engine and its systems, providing access for maintenance and engine removal, furnishing cooling air flow paths, and incorporating the necessary engine induction and exhaust systems. For preliminary study, the dual nacelle was centered vertically about the rotor chord line with the maximum diameter located approximately at the 22-percent-chord point, based upon lines taken from the profile and plan views of the wind-tunnel-model configuration. In the process of this study, the nacelle lines were further developed and improved in detail to eliminate flat areas and ramps at the engine inlets, the exhausts, and the rotor blade tip, resulting in a cleaner configuration for the wind-tunnel model.

Of primary interest at this point in the program was the location of the aft end of the nacelle relative to the engine exhaust, since it was planned that the nacelle overhang beyond the exhaust would be designed to provide an ejector to induce cooling air flow through the engine compartment. As noted previously, the initial nacelle fore and aft location was based on the wind-tunnel-model configuration and the engine center of gravity was located at the 22-percent-chord point The resultant nacelle extension beyond the engine exhaust flange was found to be approximately 12.12 inches. The maximum allowable length of airframe-installed exhaust tailpipe and nozzle aft of the engine exhaust flange was five inches, as determined from Continental's limits of snear and overhung moment for the flange. Thus, the apparent available nacelle overhang amounted to 7.12 inches. A later exhaust nozzle layout showed that the recommended nozzle area could be achieved in a nozzle length of 2.97 inches, based on the J69-T-29 exhaust cone angle, which allowed an additional two inches (approximately nine inches total) for nacelle overhang. This was considered adequate for ejector action, with the possibility that the nacelle could

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still move forward from this position and thus allow added induction system length for flow stabilization. Details of the ejector are discussed under a later paragraph entitled "Engine Cooling." It has been agreed with Continental that the exhaust nozzle can be provided with the engine instead of with the airframe as was originally planned. This will save weight, since the requirement for a flange and clamp is eliminated.

With the nacelle located as described in the foregoing paragraph, attention was next given to the engine inlet and induction system. The nacelle layout based on the wind-tunnel model showed a length of inlet duct of 11.5 inches.

Upon conclusion of the wind-tunnel tests, the results and trends were evaluated from the viewpoint of both aerodynamic and power plant installation requirements. Aerodynamic considerations based on the tests suggested that the nacelle be moved forward to an extent that a 23-inch induction system would result; but to do so would have compromised the exhaust ejector and caused engine-nacelle clearance problems. A study layout showed a compromise solution to be available if the nacelle diameter were increased to 32 inches and the nacelle were increased proportionately in length. This resulted in an inlet duct length of 17-1/2 inches and no change in fineness ratio. The selected nacelle configuration and location are shown in Figure 20, which includes the preliminary structural configuration.

Throughout the nacelle configuration development, attention was given to firewall requirements, since ability to protect the engine compartments with firewalls adequately was considered a basic requirement. The firewalls were designed to support the structure as well as to support the nacelle against aerodynamic and centrifugal loads. The firewall requirements considered involved isolation of the entire nacelle from the rotor blade, isolation of each engine compartment from the other, and isolation of each engine accessory compartment from its hot section. The selected configuration provides for these firewalls, as shown in Figure 22. The firewalls are titanium. The nacelle skin material is fiberglass from the inlet lip to station 96 frame, aluminum from station 96 to station 122.7 frame, and titanium from station 122.7 frame aft. Station 114.25 frame and all frames aft of it are titanium. Appropriate provisions are made for maintenance access, as seen in Figures 22 and 23.

3.2.5 Engine Cooling

The engine installation cooling requirements consist of lubricating oil and engine compartment cooling. The lubricating oil cooling is discussed under Section 3.2.8 entitled "Lubrication System." Compartment cooling is further broken down into accessory compartment (forward section) cooling and hot compartment (aft section) cooling. These two compartments are separated by a firewall in each engine installation.

Forward section cooling, involving low engine skin temperatures of 210 to 290 degrees Fahrenheit at 210 degrees Fahrenheit ram air temperature and involving low engine heat dissipation rates, requires only a low air flow. The required air flow is induced by pressure difference between the engine inlet duct and external nacelle pressure just forward of the intercompartment firewall. Holes are provided in the inlet duct to permit air flow between the nacelle and the duct, and in the nacelle to permit air flow in and out of the nacelle. Under static conditions, low pressure in the engine inlet will cause ambient air to flow into the nacelle through the external holes and flow through the inlet duct holes into the engine inlet. Under ram conditions, the flow will be in the opposite direction, from the inlet duct into the nacelle and out the nacelle external holes. The rotor speed will never stabilize at a value which will product a null flow point, since with only one engine of the eight operating, the rotor tip speed will be approximately 100 feet per second. The resultant ram effect, about two inches of water on a standard day, will be adequate to induce the required air flow through the forward section compartment.

The aft section cooling requirements are much more severe than those of the forward section. Engine surface temperatures at 210 degrees Fahrenheit ram air temperature vary from 500 degrees Fahrenheit just aft of the firewall to 1,300 degrees Fahrenheit at the exhaust nozzle with the latest engine configuration. Accordingly, it was considered desirable early in the program to make an analytical study of the thermal problems in the aft section, both from the standpoint of cooling and from the standpoint of determining the required nacelle material to be used in that area. The report of that analysis, pertaining to the original engine configuration, appears in Section 7.0, which shows that a 3/4-inch insulating blanket of six pounds per cubic foot density would be required on the engine aft section in order to hold nacelle temperatures in static operation to a maximum of 250 degrees Fahrenheit. However, the use of such material is considered to be entirely incompatible with a 235g environment.

The other two solutions investigated therein for maintaining a 250-degree-Fahrenheit nacelle skin, involved shrouds These were incompatible with nacelle size, and would appreciably increase the complexity of the installation. Therefore, it was decided to use a titanium nacelle in the aft section and tolerate the skin temperature which is experienced in the static condition.

Actually, the skin temperature would not reach the 800 degrees Fahrenheit indicated in Section 7.0 in the static condition, because the nacelle heat loss to atmosphere at that temperature would be approximately 40 percent greater than the heat transferred from the engine (see Section 7.0). When the rotor is rotating at 30 r.p.m. (three engines operating out of eight), the nacelle heat loss to atmosphere at 800 degrees Fahrenheit skin temperature would be ten times that transferred from the engine,

and heat transfer theory shows that under these conditions, the skin temperature would be less than 300 degrees Fahrenheit.

Section 7.0 also treats ejector theory and performance and provides parametric curves from which to determine an ejector configuration for pumping air through the aft section compartment. A positive flow of cooling air through the section is considered necessary to maintain reasonable temperatures for installation components other than the engine. Since an ejector with a 1.1 diameter ratio and a 79-percent secondary duct overhang provides adequate air flow with minimum thrust loss in hover, the ejector has been designed to these values. It is recognized that ejectors invariably require development, and it is planned that this ejector will be optimized in a later phase of the program.

In discussing cooling, it is worthwhile to discuss briefly the engine mount structure temperature. Continental has provided estimated maximum engine surface temperatures. In the area of the forward, primary mount lugs, the engine temperature does not exceed 525 degrees Fahrenheit, the radiation to the mount is negligible, and the heat conduction impediments between the engine and the mount structure (air films and necked-down sections) prevent significant conductive heat transfer. At the aft lug, where the surface temperature may be 900 degrees Fahrenheit, the same may be said in regard to conductive heat transfer, but the radiation heat transfer from the engine to the aft mount structure could be significant. If required, a small radiation shield can be added between the engine and the mount.

3.2.6 Engine Controls

The usual free-turbine-powered helicopter engine control system performs essentially three functions:

a) Provides for pilot selection of "off," "idle," or "run" setting of the gas producer.

b) Automatically increases the power turbine speed setting slightly as power demand is increased, in order to compensate for power turbine governor droop.

c) Provides the pilot with a means of making small manual in-flight adjustments of power turbine speed setting.

The tip turbojet-powered helicopter is regarded as being analogous to a free-turbine engine in that it has a gas producer (the tip engine) which supplies hot gas to drive a power turbine (the main rotor) at some governed speed irrespective of load demand. As load demand (rotor pitch) changes, the main rotor speed changes and a governor is required to sense this change and to inform the gas producer that a different power

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is required. The gas producer itself usually requires only "off," "idle," and "run" settings; but because of the need to balance engine thrusts in a multiple-engine installation, it is necessary in the tip turbojet installation to provide a method of trim for balancing engines. Early in the parametric study program, two methods of control were considered:

a) Connect the pilot's collective lever to the engine throttles. An increase in collective lever position would increase engine speed. This would tend to increase main rotor speed, and the main rotor governor would then initiate an increase in pitch to keep the rotor at governed speed. A decrease in collective lever (load) demand would decrease engine speed, the main rotor speed would tend to decrease, and its governor (sensing the decrease in speed) would initiate a reduction in pitch, to keep the rotor at governed speed. Thus, the pitch change would be accomplished as a reaction to the results of a manual input, rather than be the direct result of the input.

b) Connect the pilot's collective lever directly to the rotor pitch mechanism. The rotor governor would sense rotor speed variations and initiate changes of engine thrust to return the rotor to governed speed.

These two alternative approaches were evaluated from the viewpoints of primary flight control reliability, response time, and system complexity. Alternative a) seemed to offer a relatively rapid response to power demand but required an override to directly control the collective pitch for autorotation and flare. Further, rotor speed change with this system would be a function of drag, resulting in questionable accuracy of control with small angles of attack, which do not appreciably change the drag. Reliability of alternative a) would be inferior to that of b), since the pitch change signal would have to travel through the engine throttle system and the rotor governor system before reaching the pitch change mechanism. The decision was made to favor reliability and control accuracy by proceeding with alternative b) and to minimize response time by incorporating a bias system from the aircraft collective lever directly to the engine throttles.

With this basic decision made, a study of the devices and features, manual and automatic, required to control the tip turbine engine thrust was initiated. A question which arose early in the study involved the capability of a pilot and/or flight engineer to operate and monitor eight engines. The only known aircraft with eight turbine engines was the B-52. Discussions with Boeing Airplane Division revealed the following, gained from their B-52 experience:

a) In the B-52, the pilot has sole control of eight throttles on take-off and landing.

b) The copilot has capability of trimming engines.

c) Fuel control tolerances and small differences in fuel consumption preclude balancing engines closely by throttle relative alignments. Therefore, balance is achieved by comparison of exhaust gas pressure ratio.

d) Automatic engine control is necessary in the tip turbojet application.

This information, combined with conclusions reached at Hiller Aircraft Company, led to a concept of tip turbojet engine control on which the system described herein is based.

The resultant power management system enables the pilot to control and coordinate all operating engines with a minimum of manual inputs to the system. It contains fail-safe features and backup systems consistent with the high degree of safety and controllability demanded of primary flight controls. The systems, both basic and backup, are combined electronic and electro-mechanical. Figure 24 is a block diagram of the complete system. There are three methods of control operation: manual, automatic, and mixed. Manual is used for engine start and shutdown, static ground run, and as a backup for the automatic system. The automatic system is used in normal operation. A combination of manual and automatic is used to permit manual operation of a portion of the automatic system in case of partial automatic failure. In all regimes, fine-trim balancing of individual engines is accomplished with the throttles.

In manual operation, major power changes are made by a device actuated by the collective lever, the power change being proportional to the amount of collective lever movement. Provisions for trim, to balance power available to power required, and to maintain desired rotor speed, is accomplished by a twist grip mounted on the collective lever. The requirement for this manual trim arises from the variation in power required versus collective lever position for varying gross weights, altitudes, and forward speeds. This manual collective lever power coordination and twist-grip trim is similar to the usual reciprocating engine-powered helicopter engine control.

In automatic operation, the main rotor governor automatically senses rotor speed changes resulting from load changes, and informs the engine fuel controls of the need for thrust change. In order to minimize the main rotor transient speed change occurring during the time in which the governor and engine fuel controls are acting to meet the required change in thrust, an anticipating device is incorporated in the system. This device, actuated by the collective lever, sends a prior signal of anticipated thrust change to the engine fuel controls, so that a change

in thrust has already been initiated by the time the main rotor governor signal reaches the fuel controls.

The mixed operation feature of the power management system provides for manual operation of one or more engines while the other engines are being automatically operated. In this mode of operation, main rotor speed is held constant by the rotor speed governor, as in the automatic mode, and collective stick position and manual trim are used to match the thrust of the engine or engines in the manual mode with the thrust of the remaining engines which are in automatic mode.

An individual three-setting throttle lever is provided for each engine in the cabin throttle group. Each throttle has STOP and IDLE-START detent positions and a RUN range. The detent positions are used only in manual operation, and the RUN range is used for all conditions of operation. Relative thrust adjustments between engines may be made during flight by movement of throttles within the RUN range to allow all engines to approach their operational limits uniformly. During static ground running, the throttle is used in the manual mode to control the engine throughout its full power range.

Individual mode selector switches are included for each engine throttle, providing for selection of MANUAL or AUTOMATIC operation. In addition, a two-position master mode selector switch is incorporated, with the two positions designated SELECT and MANUAL. In the SELECT position, the master mode selector switch permits each engine to be operated in the mode called for by its respective mode selector switch position. In the MANUAL position, the master mode selector switch overrides the individual mode selector switches to place all engines in the manual mode.

A rotor speed selector is used to select the rotor governor speed setting in the automatic and mixed modes. Rotor speed is held within three percent of governor speed setting at all times; however, a warning is provided at five percent underspeed or overspeed to initiate corrective action by the pilot in the event of a failure.

All thrust change requirements in either mode of operation ultimately result in a movement of an input lever on the engine fuel control. Dual actuators are provided for this input motion, one for manual mode and one for automatic mode. Each actuator of a pair can provide control throughout the full thrust range of the engine, irrespective of the position of the other actuator, the particular actuator in use depending upon whether manual or automatic mode is being used.

It is anticipated that the tip turbojet flight article may require provision for smoothing out oscillatory engine thrust and rotor drag variations which may be encountered due to advancing and retreating blade effects in forward flight, depending upon the control capabilities of the

basic engine fuel control. Provisions are made to incorporate a cycling power control into the existing power management system if it is warranted.

Section 5.1 of this report presents the Engine Power Management Specification which was prepared by Hiller and distributed to qualified vendors for cost estimates. In this specification, vendors were asked to propose methods for automatic thrust equalization between engines, as well as oscillatory thrust variation control mentioned in the above paragraph. Vendor replies confirm that the system is feasible and practicable in its entirety and have suggested at least two ways of automatically controlling thrust equalization. These are

a) Comparison of engine rotor r.p.m.
b) Comparison of tailpipe temperature.

3.2.7 Fuel System

The fuel system discussed in the following paragraphs of this section treats all elements of the system except the fuel tank itself.

The first study undertaken in this system was to determine the pressures to which the fuel lines and components would be subjected, particularly those pressures created by centrifugal force acting on the fuel in lines in the rotor. At the time of the initial study, a 65-foot-radius rotor rotating at 105 r.p.m. was assumed, and values of fuel pressure versus distance from the rotor hub for 105 r.p.m. and for 1.25 times 105 r.p.m. (overspeed limit of 131.2 r.p.m.) were calculated. Tip pressures were calculated to be 2,645 p.s.i. at 105 r.p.m. and 4,140 p.s.i. at 131.2 r.p.m. The parametric study ultimately yielded a 56-foot-radius rotor rotating at 111 r.p.m., for which the tip fuel pressures were 2,200 p.s.i. and 3,490 p.s.i. at rated and overspeed r.p.m., respectively.

The maximum required fuel pressure at the engine fuel control at that time was quoted by Continental as 120 p.s.i. at full power. Therefore, fuel pressure at the rotor tip would vary from 120 p.s.i. for the single engine static ground run condition to 3,470 p.s.i. at 125 percent of rated rotor speed. The pressure at the rotor hub, on the other hand, would have to be some value higher than 120 p.s.i. for the static ground run condition, to allow for line pressure drop and would have to be at only a very small pressure for nonstatic conditions since the centrifugal force will pump the fuel from the hub to the tip. The required fuel pressure at the engine fuel control was subsequently reduced to 80 p.s.i.g.

An analysis was conducted of fuel line size required to minimize fuel pressure drop in the static ground run condition. This was desirable to alleviate the problems associated with transferring the fuel from the aircraft into the rotating rotor mast, and to minimize fuel pump require-

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ments. The analysis showed a requirement for 3/4-inch tubing to satisfy the static condition. Based on 3/4-inch tubing and the calculated pressure data, the required tubing strength and resultant wall thickness was calculated for various points on the rotor radius. From this, a system of increasing tube wall thickness versus distance out on the rotor radius was designed.

Subsequent to this work, several design decisions pertinent to the rotor fuel system were evolved from the parametric study:

a) An over-under engine arrangement will be used.

b) The rotor blade radius to the engine centerline will be 56 feet.

c) The rotor blade may require some ballast forward of the feathering axis, based on the axis location at the 25-percent-chord point.

d) Rotor speed will be 111 r.p.m.

As noted previously, the maximum rotor tip fuel pressures resulting from a 56-foot-radius rotor were 2,200 p.s.i. at 111 r.p.m. and 3,490 p.s.i. at 125 percent of 111 r.p.m. These pressures are based on JP-4 fuel at 60 degrees Fahrenheit. As a check, a calculation was made of the overspeed fuel pressure which would exist with JP-5 fuel, which is heavier, at -60 degrees Fahrenheit. This resulted in a value of 3,910 p.s.i.

The previous fuel line design had been based on the use of 3/4-inch aluminum tubing of varying wall thickness. In view of the possibility of forward blade ballast being required, it was decided to optimize the fuel line design for factors other than weight. Accordingly, a titanium line of constant wall thickness was designed, so as to provide the same modulus of elasticity and coefficient of expansion as the titanium leading edge spar to which it is attached. In this design, a factor of 2.5 times operating pressure was used as in the hydraulic system design, because the normal fuel system design factor of four times operating pressure (intended primarily for relatively low pressure systems) was considered not applicable and unrealistic. The resulting fuel line size was a 3/4-inch tube of .058-inch wall thickness.

Figure 25 shows the aircraft fuel system in schematic form, from which the following features are evident:

a) Each engine is supplied with a separate fuel line from the rotor hub to the engine.

b) Each engine fuel supply line has a shutoff valve at the rotor hub.

c) Each engine fuel supply line has a flow limiter at the rotor hub.

One 3/4-inch fuel line through the rotor would be adequate to supply the static ground run fuel requirements of one engine and the normal operational fuel requirements of two engines; however, it would then be necessary to provide a tee at the rotor tip with lines leading to each engine. This would require that, in case of a fuel leak or a fire at the engine, either the fuel to both engines would have to be shut off by a single valve at the hub or two valves would have to be located at the rotor tip to shut off fuel to an individual engine.

Reliability demanded that the system provide for shutting off the fuel to individual engines without affecting the other engines; hence, there will be two fuel shutoff valves per rotor blade, one per engine.

Analysis showed that the effective weight of a single fuel line running out the rotor blade and two valves located at the rotor tip (the actual component weight plus the weight of structure required to support it at its location in space) would be approximately the same as the effective weight of two valves located at the rotor hub and two lines running out the rotor blade, one to each engine. Thus, for approximately the same effective weight, the following features are added if the fuel shutoff valves are located at the rotor hub:

a) Two fuel lines may be run out the rotor blade, so that in case of line failure only one engine is lost.

b) The shutoff valves will not be subjected to high pressure or operation in a high acceleration field and may be simple, off-the-shelf items. Hence, their reliability will be improved.

Based on the above analysis, the design selected consisted of an individual fuel line from the rotor hub to each engine, with a shutoff valve for each line located at the hub.

Since the rotor fuel lines are titanium, the shutoff valves serve as firewall shutoff valves. The function of the flow limiter in each fuel line at the hub is to prevent starving all engines of fuel with the pumping action of the centrifugal field should any fuel line suddenly fail completely.

As shown in Figure 25, two identical, independent, but interconnected fuel systems feed the two manifolds at the rotor hub which serve the eight engines.

The upper manifold at the hub serves the upper engine on each of the four rotor blades; the lower manifold at the hub serves the lower engine on each rotor blade. Each manifold is supplied by a separate fuel system, consisting of a tank-mounted electric boost pump, thence through a filter and a shutoff valve. A check valve is located immediately

outside the tank to prevent draining the fuel in the system back to the tank upon shutdown, and the shutoff valve is located immediately downstream of the filter to permit filter cleaning without draining the fuel out of the rotor mast.

Each fuel system feeds through a sealed joint into a rotating manifold near the bottom of the rotor mast. This is discussed elsewhere in this volume and shown in Figure 12. From there, the fuel is led up the inside of the rotor mast to the upper and lower distribution manifolds, also shown in Figure 12. The two manifolds are interconnected through a shutoff valve so that dual fuel system reliability exists from the fuel source to the manifolds, with dual pump reliability in each system.

Fuel pump size was based on two conditions, static ground run maximum power of one engine and operational maximum power of eight engines. Under all conditions, a fuel pressure of at least 80 p.s.i.g. is required at the engine fuel control inlet. The maximum fuel requirement of a single engine at static conditions at 1,700 pounds thrust is 4.7 g.p.m. at 80 p.s.i.g. at the engine fuel control inlet. Allowing for system pressure drop, 100 p.s.i.g. is required at the pump outlet. Under operational conditions, with maximum power at maximum forward flight speed, and with one fuel pump system serving all eight engines, a total of 41.2 g.p.m. capacity is required by the engines Under these conditions, however, only 10 p.s.i.g. pump pressure is required to overcome the losses to the top of the rotor mast; from there, the fuel is pumped to the engine fuel controls by centrifugal force. (The fuel control also has a provision for depressurizing the fuel under operational conditions.) The resultant fuel pump requirements to which each electrical boost pump and each mechanical boost is designed, therefore, are 5 g.p.m. at 100 p.s.i.g. and 52 g.p.m. at 10 p.s.i.g. Note that for the operational condition, a normal FAA fuel system capacity design factor of 1.25 has been applied.

3.2.8 Lubrication System

The Continental model 357-1 engine used in this study has a self-contained oil tank. The engine oil tank, consisting of an annular volume within the engine case between the axial and centrifugal compressor stages, carries enough oil to supply the engine lubrication requirements including the specified maximum oil consumption of 1.0 pound per hour, for the aircraft mission defined in Reference 18. Oil replenishment requirements, if any, were to be determined by Hiller, and replenishment capability was to be provided by Continental. In addition, oil cooling requirements were to be investigated and cooling accomplished by Continental.

It was envisioned initially that oil replenishment might not be required, in view of the engine tank capacity and the moderate consumption rate.

As the aircraft characteristics began to evolve, it became apparent that field checking of oil level at the rotor tip would be difficult; also, a variable and unpredictable imbalance between rotors because of oil leaks or excessive oil consumption at one rotor tip would be undesirable. Accordingly, it was concluded that an oil replenishment reservoir should be provided in the aircraft, in an area unaffected by centrifugal force and readily accessible. This arrangement would provide a level of reliability considered essential and would eliminate the weight of oil reserves carried at the rotor tips (with the attendant elimination of weight of structure required to support the oil reserves at the tips). Five different configurations of replenishment system were considered:

a) A simple reservoir, located at the foot of the rotor mast, with a continuously operating pump, with bypass, to maintain a constant head of oil at the rotor hub. Centrifugal force would then maintain a head from the rotor hub to the rotor tips, except for the short-duration static ground run condition during which replenishment would not be required.

b) A single reservoir located at the foot of the rotor mast, with a pump actuated by signals from the engine low-level switches.

c) Two configurations involving the use of hydraulic system pressure to pressurize the replenishment oil, with a single reservoir at the foot of the rotor.

d) A reservoir located in the root of each rotor blade, with consequent elimination of need for a pump or for oil to travel through the rotor mast.

e) Use of the aircraft accessory gearcase oil system as a source of oil and pressure.

Alternative e) was eliminated on the basis of unreliability, since it would result in probable contamination of engine oil should a gearcase failure occur. Of the remaining alternatives, the best combination of reliability, light weight and simplicity was found to be in the system with the continuously operating pump and bypass. The oil replenishment system is shown schematically in Figure 26. For weight considerations, aluminum lines are used to route the oil through the rotor blades.

Oil cooling means were planned to be provided as part of the engine, but study showed that an engine-mounted oil cooler would require enlarging of the nacelle frontal area to accommodate the cooler and cooling air ducting. Therefore, attention was directed toward skin-type coolers, which could use the high air velocities past the nacelles for highly effective convective cooling. However, close study and analysis revealed the following disadvantages in this approach:

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a) In spite of the high air velocity past the nacelle, the required surface area of a simple skin cooler would necessitate the use of fins, with attendant drag losses.

b) Considerable tooling would be required to manufacture such a cooler and would be wasted if resizing of the cooler were found necessary after initial testing.

A modification of the skin cooling concept was devised, involving use of a conventional oil-air heat exchanger submerged in the rotor blade. The configuration is shown in Figure 27. Under normal operating conditions, boundary layer air flows into the heat exchanger through turning vanes from the under side of the rotor, passes through the heat exchanger, and exits out the top of the rotor blade through turning vanes which deflect it aft. A portable, electrically-driven blower is used to provide air for static ground run.

The heat exchanger is sized for conditions of hover in ground effect at sea level on a 115-degree-Fahrenheit day. Under these conditions, the turbine temperatures are a maximum, oil heat rejection is a maximum of 1,200 BTU per minute per engine, a 60-degree-Fahrenheit ram temperature rise will exist, and an exhaust recirculation temperature rise of 10 degrees Fahrenheit is assumed to exist. Using the resultant design air inlet temperature of 185 degrees Fahrenheit to the cooler, a duplex cooler core of 16-inch length, 13-inch width, and 3.75-inch depth was found to meet the oil cooling requirements of two engines.

An oil cooler design summary is included in Section 6.0 (Appendix II). The cooler is mounted in an almost flat position in the aft portion of the rotor blade, near the tip. The forward edge of the cooler is at the 56-percent-chord point, and the centerline, inclined 6-1/3 degrees upward from the rotor chord, is located at rotor blade station 658.65. A ramped, vaned inlet is located on the bottom surface of the rotor blade, and a vaned flush exit turns and accelerates the air leaving the cooler at the top of the rotor blade to give maximum recovery. Because heat rejection increases as power (rotor pitch) increases, and pressure ratio across the blade (hence pressure ratio across the cooler) increases as rotor pitch increases, the system inherently offers a balancing influence between cooling required and cooling available. Additionally, it has been studied aerodynamically and found to be completely acceptable from an aerodynamic viewpoint.

For static ground run operation for maintenance purposes, the inlet and exit vane assemblies are hinged, so that they may be swung aside and an electrically-driven portable blower may be mounted directly to the cooler on the under side of the rotor. A socket is provided in the cooler well, for operation of the blower from the aircraft's 115-volt, 400-cycle, single-phase AC power system.

3.2.9 Starting System

Engine starting received extensive study, both by Hiller and by Continental, to assure that the starting system which emerged would be practicable, efficient, and optimum for the purpose. The methods of engine starting considered were:

a) Electrical cranking

b) Hydraulic cranking

c) Cartridge cranking

d) Windmill starting by main rotor cranking

e) Air impingement cranking

Simultaneously, consideration was given to the minimum number of engines for which starting would have to be provided, with the thought that the balance of the engines could be started by windmilling after the rotor had begun to turn. Since the required rotor speed range for windmilling would depend on engine windmill-starting characteristics, and the rotor speed for a particular quantity of engines idling would depend on engine idle thrust and rotor inertia, the necessary investigation into these questions was initiated and primary attention was given to the mode of starting to be used.

For purposes of studying the mode of starting, it was assumed that only one engine per blade would be cranked, and that the other engine would be windmill-started; further, that each engine would have to be capable of starting by some means for static ground run operation. A summary of the study findings follows.

a) Electrical cranking was disqualified on the basis of the high weight of the starters and wiring in the rotor and at the tip; and the necessity for transferring large amounts of electrical power from the fuselage to the rotor mast through slip rings. One engine at each rotor tip would mount a 25-pound starter; the other engine would require either a similar starter or other means for static ground run starting. All engines would carry the accessory case weight necessary to mount and support a 25-pound starter, and each rotor blade would have to carry approximately 14 pounds of size number 2 electrical wire. The weight of the slip rings involved would be relatively small (approximately three pounds), but the reliability requirements of the slip rings in transmitting several hundred amperes starting current would offer maintenance problems.

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b) Hydraulic cranking would involve only about 10 pounds motor weight per engine at the rotor tip, but would require hydraulic motors on all engines to provide for the static ground run condition. The principal drawback in a hydraulic system is the problem of bringing 3,000 p.s.i. hydraulic pressure from the fuselage into the rotating rotor mast and then out through the rotor blades to the engines. In addition, there would be a requirement for, and added weight of, a clutching device operating in the high g field to disconnect the hydraulic starting motor from the engine. Hydraulic cranking was disqualified on the basis of excessive complexity and rotor tip weight and inferior reliability relative to the other systems considered.

c) Cartridge starting presents problems which are considered incompatible with the helicopter, and is consequently rejected as a starting method. The tip weight of a cartridge-starting·assembly would be approximately 40 pounds per engine. Reloading with fresh cartridges would necessitate a ladder to reach the nacelle and provision for access.

d) The least complex and most straightforward method appears to be windmill starting by central cranking of the helicopter rotor. Unfortunately, this system presents problems from a weight standpoint. Figure 28 shows the range of rotor tip speeds over which windmill starts can be accomplished. The parametric study rotor system, with four blades and eight engines, has an inertia of $750,000$ slug-feet^2, for which it is calculated that 445 horsepower would be required at the rotor mast to reach engine windmilling light-off speed in one minute.

Preliminary study indicated that the gearing and overrunning clutch required to impart 400 - 500 horsepower to the rotor mast at its slow rotational speed would add at least 250 pounds to the weight of the aircraft. In view of this considerable weight penalty, it was decided to disqualify the rotor cranking system. However, it is recognized that this system offers many advantages, and attention to it will be maintained in order that it may be reconsidered should the ground or flight test vehicles of the future involve design changes which would permit its use without so severe a weight penalty. Ground start capability could still be accomplished by air impingement.

e) The four starting modes above were considered to be inferior as compared to the system selected, air impingement starting. In this method, the engine is cranked to its starting speed by directing compressed air into the axial stage of its compressor. The capability for air impingement starting will be incorporated on the upper engine on each blade, with the compressed air routed

from the fuselage-mounted APU into the rotor mast, up the mast, and then out the rotor blades to the engines. The lower engine on each blade will be started by windmilling. All eight engines will have the capability of being ir-impingement started from a ground starting cart for static operation.

The required engine impingement flow of 1.25 pounds per second at 70 p.s.i.a. is supplied by the aircraft auxiliary power unit, which has a capability of delivering 2.0 pounds per second at 130 p.s.i.g. at standard conditions, thus allowing margin for nonstandard conditions, pressure drop, and heat loss between the APU and the engines.

The starting air is ducted from the APU to the rotor through a flexible duct, passes through a rotary manifold into the rotor mast, and travels up the rotor mast to a manifold that has four individual valves for selecting the engine to be started. From the valve, the air is ducted through a three-inch duct out each rotor blade to the upper engine at the blade tip.

In the starting sequence, one upper engine is started followed by its counterpart on the opposite rotor blade, and then by the remaining two upper engines, one at a time. The throttles are then advanced to bring rotor tip speed up to 254 feet per second (the minimum windmilling light-off speed shown in Figure 28), and the lower engines are then started in order. As seen from Figure 28, the resultant tip speed with all engines idling does not exceed the maximum light-off tip speed of 338 feet per second. The estimated weight of the complete system, except for the APU, is 172 pounds.

As noted previously in this report, the APU provides a power source for accessory and tail rotor drive and ground checkout, and its weight is justified on that basis. A schematic diagram of the engine starting system is shown in Figure 29.

3.2.10 Fire Safety

Some discussion of fire safety of the power plant installation has been presented in previous paragraphs. In the nacelle discussion, it was pointed out that the engine compartment firewall, as shown in Figure 22, isolates each engine from its neighboring engine and from the main rotor blade; in addition, transverse firewalls isolate each engine's compressor section from its combustor, turbine, and exhaust sections. The fuel supply line to each engine is fireproof material (titanium) from the hub-located shutoff valve to the engine. The oil replenishment line for each engine is supplied with a shutoff valve at the rotor tip. Thus, the rotor structure and the main structure are completely isolated from the fire zones.

In the event of a fluid line or fitting failure in flight, centrifugal force would throw the fluid outward, away from the aircraft. Drain holes will be provided at the rotor tip so that, should a failure occur. in the rotor-located portion of any fluid system, the fluid will be drained from the rotor. Since no airframe flammable fluid line passes into or through the engine hot section compartment, fluid line failure will not result in exposure of flammable fluids to any ignition source. Any fluid leakage in the engine accessory compartment will be thrown out of the nacelle cooling air inlet holes, by centrifugal force.

Fire detection provisions are incorporated in the power plant installation, to give warning to the pilot to shut off the flow of flammable fluids to the affected area. Fire extinguishment provisions are not incorporated, since no concentration of leaked flammable fluids can occur within the nacelle during normal operation. During static ground run, ground extinguishing equipment will be available.

The fire detection system selected is a photo-electric type employing no moving parts in the detector. It provides 100-percent volume surveilance of the fire zone, is simple to install, requires no shock mounts, and has an approximate 10,000-hour life expectancy.

3.3 Electrical System

3.3.1 Power Distribution System

The power distribution system is supplied by two separate sources, as shown in Figure 30. The auxiliary power unit (APU) is the primary source and the main rotor accessory gearcase is the secondary source. Harness routing and electrical system disconnects are shown in Figure 31.

The APU provides DC power from a starter generator and emergency DC power from its battery. Two 30-kilovolt-ampere, 6,000-r.p.m., 400-cycle generators are driven directly by the APU gearbox. As noted in Figure 30, one generator supplies alternating current to the main bus, and the other generator supplies alternating current to the essential bus, with a switch-operated power relay between the two at the bus bar. This system is considered less complicated than a parallel circuit to the bus. The use of two generators, in preference to a single 60 KVA generator, provides a greater degree of reliability.

The secondary electrical source consists of a 15-KVA, 6,000-r.p.m., 400-cycle generator driven from the main rotor accessory gearbox. This system provides alternating current to the essential bus and from there to the primary flight electrical equipment. If a failure occurs in the APU, this system will provide the necessary power for completing flight to the nearest airfield. In the secondary system,

consideration will be given either to a two-speed gearbox or a constant-speed drive unit between the main rotor gearbox and the generator, to provide for constant generator speed if different rotor speeds are used for cruise and hover. Final selection will be evaluated in detail design, which is not within the scope of this study.

External power receptacles have been provided for both AC and DC circuits so that ground checkouts of various systems can be accomplished.

3.3.2 Slip Rings

The slip ring assembly is a key item in the electrical distribution system. Approximately 320 rings are required, located as shown in Figure 12. Thorough investigation has indicated that a modular platter-type ass 1bly will give best reliability, low noise level, and long service life. This type has the rings arranged concentrically on the tops and on the bottoms of a series of disks, with two or more silver-graphite block brushes per ring. The assembly will be designed for a minimum of 500 hours without attention.

3.3.3 Power Management

The functions of the power management system are described in Section 3.2.6 of this report. The power management system block diagram is shown in Figure 24.

3.3.4 Power Plant Instrumentation System

The engine monitoring system shown in Figure 16 has two basic systems: a) the primary, which is concerned with engine shaft speeds and turbine inlet temperature, and b) the secondary, which monitors oil temperature, oil pressure, and fuel pressure. The components of these systems which operate in the high g environment have been extensively discussed with several prominent manufacturers. This has led to the conclusion that existing standard units, namely, the rotary tachometer generators, pressure synchros, and resistance-type temperature bulbs, can be used without modification. Early centrifuge testing in the next program phase will determine the validity of this conclusion.

As shown in Figure 16, each tip turbojet is provided with a group of sensors or transmitters, with the exception of fuel pressure units. The fuel pressure is measured at each fuel system manifold located at the top of the rotor mast. It is considered unnecessary to indicate pressures at the engines because of the very large pressure increment provided by centrifugal force.

The primary system includes the following:

a) Tachometer system

b) Turbine inlet temperature system

The tachometer system is of standard indicator and rotary generator design. The indicator reading is in percent with a range of 0 to 115 percent, adjusted to indicate 100 percent when the generator speed is 4,200 r.p.m. The turbine inlet temperature system is a servo type to provide best available accuracy and reliability for this important parameter. The indicators read in degrees centigrade with a range of 0 to 1,000°C. A caution light is included to indicate the approach of the maximum allowable turbine inlet temperature.

The secondary system includes the following:

a) Oil temperature system

b) Oil pressure system

c) Fuel pressure system

The oil temperature system is a standard resistance bulb-type incorporating an MS28034-3 bulb and an MS28009-2 type indicator. The indicator range, however, is -20°C. to +130°C. for added indicator readability. The oil and fuel pressure systems are of synchro type and the indicators read in p.s.i. with a 0- to 200-p.s.i. range for the oil pressure and 0 to 150 p.s.i. for the fuel pressure. The cockpit instrument arrangement is shown in Figure 17.

3.3.5 Auxiliary Power Unit Instrumentation

These instrument systems, shown in Figure 32, include the following:

a) Tachometer system

b) Oil temperature system

c) Oil pressure system

d) Turbine inlet temperature system

All the above systems consist of instruments of the same basic design as those used for the engine with the exception of the turbine inlet temperature instrument system where a direct reading thermocouple system is employed.

STUB BLADE

GIMBAL UNIVERSAL
POINT (W.L. 400.00)

ROTOR SHAFT

ENGINE OIL, FUEL, AIR,
AND ELECTRICAL OUTLETS
TO ROTOR BLADES

1108-510
ROTOR HUB ASSEMBLY

1108-530
MAIN ROTOR BLA
ASSEMBLY

Figure 1.

ROTOR THRUST BEARING —
MOUNT

AUXILIARY FLIGHT
ACCESSORIES GEARCASE

— ROTOR SPRING
RESTRAINT SYSTEM

— ALTERNATE T

UPPER MASTER —
SUPPORT BEARING
W.L. 338.75

W.L. 316.00

W.L. 286.00

— 1108-300
MAIN ROTOR FLIGHT
CONTROLS INSTALLATION

— TANDEM HYDRAULIC BOOST
CYLINDER (3 PLACES)

—1108-530
MAIN ROTOR BLADE
ASSEMBLY

SIDE VIEW

← UP

↓ FWD

Figure 1. Rotor System Assembly - Tip Turbojet.

B

AUXILIARY FLIGHT
ACCESSORIES GEARCASE

ALTERNATE TAIL ROTOR DRIVE

W.L. 286.00

ENGINE STARTING
AIR DUCT FROM
AUXILIARY POWER UNIT

POWERPLANT SYSTEMS ELECTRICAL
SLIP RING ASSEMBLY

ENGINE OIL INLET
SWIVEL

DUEL FUEL
MANIFOLD

1108-230
MAIN ROTOR TRANSFER
SYSTEM ASSEMBLY

NDEM HYDRAULIC BOOST
LINDER (3 PLACES)

UP

FWD

et.

C

GIMBAL RING

ADJUSTABLE
LINK ASSY

HUB

30 IN.

ISOLATI
(RE

PIVOT
PIN

ROTOR
ATTACH
PIN A

TIMKEN ROLLER
BRG
CUP -NO. 772
CONE -NO. 787

GIMBAL RING

STUB
BLADE

15.75
IN.

PIVOT
BRGS

15 IN.
DIA

ATTACHM
LUGS

32.50
IN DIA

ROTOR
MAST

RESTRAINT
SPRINGS

VIEW A-A

RESTRAINT SPRING
COLLAR

SECTION B-B

Figure 2. Main Rotor

STUB BLADE

1108-530
MAIN ROTOR BLADE ASSEMBLY

ATION LINK
REF)

TENSION-TORSION
STRAP ASSEMBLY

R BLADE
CHMENT
ASSY

IMENT

SECTION C-C

TENSION-TORSION
STRAP ASSEMBLY

tor Hub Assembly.

B

INTERNAL STRAP

EXTERNAL STRAP

MODIFIED INTERNAL STRAP

Figure 5. Hub to Blade Retention Systems, Sheet 1.

47

FLEXURAL HINGE

LAMINATED BEARINGS

STUB BLADE

Figure 4. Hub to Blade Retention Systems, Sheet 2.

48

STUB BLADE LEADING EDGE

.12 THICK SKIN
TI-8AL-1MO-1V TITANIUM

DRAG LINK ATTACHMENT

BLADE TO HUB
ATTACHMENT
LUG

TENSION-
TORSION
STRAP
(REF)

6.5 DIA. PIN

FEATHERING AXIS

PITCH CONTROL HORN

BLADE TRAILING EDGE

.0020 x 3/8 5052 ALUMINUM ALLOY
FULL DEPTH CORE

VIEW A-A

Figure 5. Main Rotor B

.12 THICK SKIN
TI-8AL-1MO-1V TITANIUM ALLOY

6.5 DIA. PIN

REMOVABLE LEADING EDGE ACCESS COVE

B

Ti-8A

A

AF

HON
.0015 AL. FO
.020 Ti. A
.010 AL. A

.0020 x 3/8 5052 ALUMINUM ALLOY
FULL DEPTH CORE

TOP PLAN VIEW
INBOARD

B

.10 THICK SKIN
TI-8AL-1MO-1V TITANIUM ALLOY

Figure 5. Main Rotor Blade Root Retention Structure.

B

IN

VABLE LEADING EDGE ACCESS COVERS

B

NOSE SPAR AT 5% CHORD

Ti-8AL-IMO-IV TITANIUM
ALLOY PREFORMED
SKIN

FWD. CELL RIB
.032 TI ALY

AFT WEB AT 50%
CHORD

HONEYCOMB PANEL
.0015 AL.FOIL x 1/4 HEX CELL CORE
.020 TI. ALY. OUTER FACING;
.010 AL.ALY. INNER FACING)

TRAILING EDGE RIB
.032 TI ALY

78 IN. CHORD

PLAN VIEW
BOARD   B

SECTION B-B   SECTION AT TYP. RIB STA.

ALLOY

C

Figure 6. Running Weight (Basic) Versus Blade Chord.

Figure 7. Running Weight (Basic) Versus Blade Chord.

FRONT ENGINE MOUNT
PICK-UP FITTINGS

STA 648.00

VIEW A-A

REAR ENGINE MOUNT
PICK-UP FITTINGS

STA 654.50

VIEW B-B

TI-8AL-IMO-IV TITANIUM ALLOY
PREFORMED SKIN

5 % CHORD

50 % CHORD

TRAILING EDGE SKIN PANEL (.38 THICK HONEYC
.020 OUTER FACING, TI-8AL-IMO-IV TITANIUM
.010 INNER FACING, 2024-T3 ALUMINUM ALLO
.0015 X 1/4 CELL.5052 ALUMINUM ALLOY COF

TOP PLAN VIE
INBOARD

Figure 8. Engine Retention Structure

53

TI-8AL-IMO-IV TITANIUM ALLOY
PREFORMED SKIN

ENGINE NACELLE (REF)

ENGINE MOUNT
TI-6AL-6V-2S

C

BLADE FEA

ENGINE MOUNT

A

A

B

B

FAIRING ATTACHING
FASTENERS

ENGINE INSTALLATION ₵
STA 672.00

TIP CLOSING RIB ₵

STA 636.00

C

TRAILING EDGE SKIN PANEL (.38 THICK HONEYCOMB,
.020 OUTER FACING, TI-8AL-IMO-IV TITANIUM ALLOY
.010 INNER FACING, 2024-T3 ALUMINUM ALLOY
.0015 X ¼ CELL.5052 ALUMINUM ALLOY CORE)

TOP PLAN VIEW
INBOARD

Figure 8. Engine Retention Structure - Main Rotor Blade Tip.

B

ENGINE MOUNT SUPPORT FITTINGS
TI-6AL-6V-2SN TITANIUM ALLOY

C

BLADE FEATHERING AXIS

A

B

₵ FRONT ENGINE MOUNT
1108-630 (REF)

₵ REAR ENGINE MOUNT
1108-631 (REF)

AFT RIB, .050 THICK,
TI-6AL-4V TITANIUM ALLOY

AIRING ATTACHING
FASTENERS

ENGINE INSTALLATION ₵
STA 672.00

TP CLOSING RIB ₵

VIEW C-C

C

C

TI-8AL-1MO-1V TITANIUM ALLOY
.12 THICK SKIN

STUB BLADE

OUTBOARD RETENTION PIN

78.0 INCHES
(CONSTANT)

INBOARD RETENTION PIN

STA 0.00

TRAILING

ROTOR HUB (REF)

₵ ROTOR SHAFT

1108-531 MAIN ROTOR BLADE
ROOT RETENTION STRUCTURE

SKIN
TI-8AL-IMO-IV TITANIUM ALLOY
(THICKNESS TAPER: .14 AT ROOT TO .10 AT TIP)

NOSE SPAR ₵ (5 % CHORD)

BLADE

REMOVABLE LEADING EDGE SEGMENTS
(.10 THICK TI-6AL-4V TITANIUM ALLOY)

TRAILING E[
.020 OUTER
.010 INNER
.0015 X ¼

F[
(₂

RIB ₵ (TYPICAL)

REAR SPAR ₵ (50 % CHORD)

TIP CLO

593.0 INCHES

PLAN VIEW

1108-710 ENGINE INSTALLATI

TRACE OF CHORD SURFACE AT 25 % CHORD
(BLADE TWIST: 10° AT ₵ TO 0° AT STA 672.00)
AIRFOIL SECTION: MODIFIED N.A.S.A. 0015

.ING EDGE

LEADING EDGE

1108-532
MAIN ROTOR BLADE TIP
ENGINE RETENTION STRUCTURE

INBOARD

Figure 9. Main Rotor Blade Assembly.

B

) .10 AT TIP)

€ (5 % CHORD)

TRAILING EDGE SKIN PANEL (.38 THICK HONEYCOMB,
.020 OUTER FACING , TI-8AL-IMO-IV, TITANIUM ALY.
.010 INNER FACING, 2024-T3 ALUMINUM ALLOY
.0015 X 1/4 CELL 5052 ALUMINUM ALLOY CORE)

REMOVABLE LEADING EDGE SEGMENTS
(.10 THICK TI-6AL-4V TITANIUM ALLOY)

FEATHERING AXIS
(25% CHORD)

A: € (50 % CHORD)

TIP CLOSING RIB

STA 672.00

1108-710 ENGINE INSTALLATION (REF)

TRACE OF CHORD SURFACE AT 25 % CHORD
(BLADE TWIST: 10° AT € TO 0° AT STA 672.00)
AIRFOIL SECTION: MODIFIED N.A.S.A. 0015

1108-532
MAIN ROTOR BLADE TIP
ENGINE RETENTION STRUCTURE

C

Figure 10.   Specific Modulus of Elasticity.
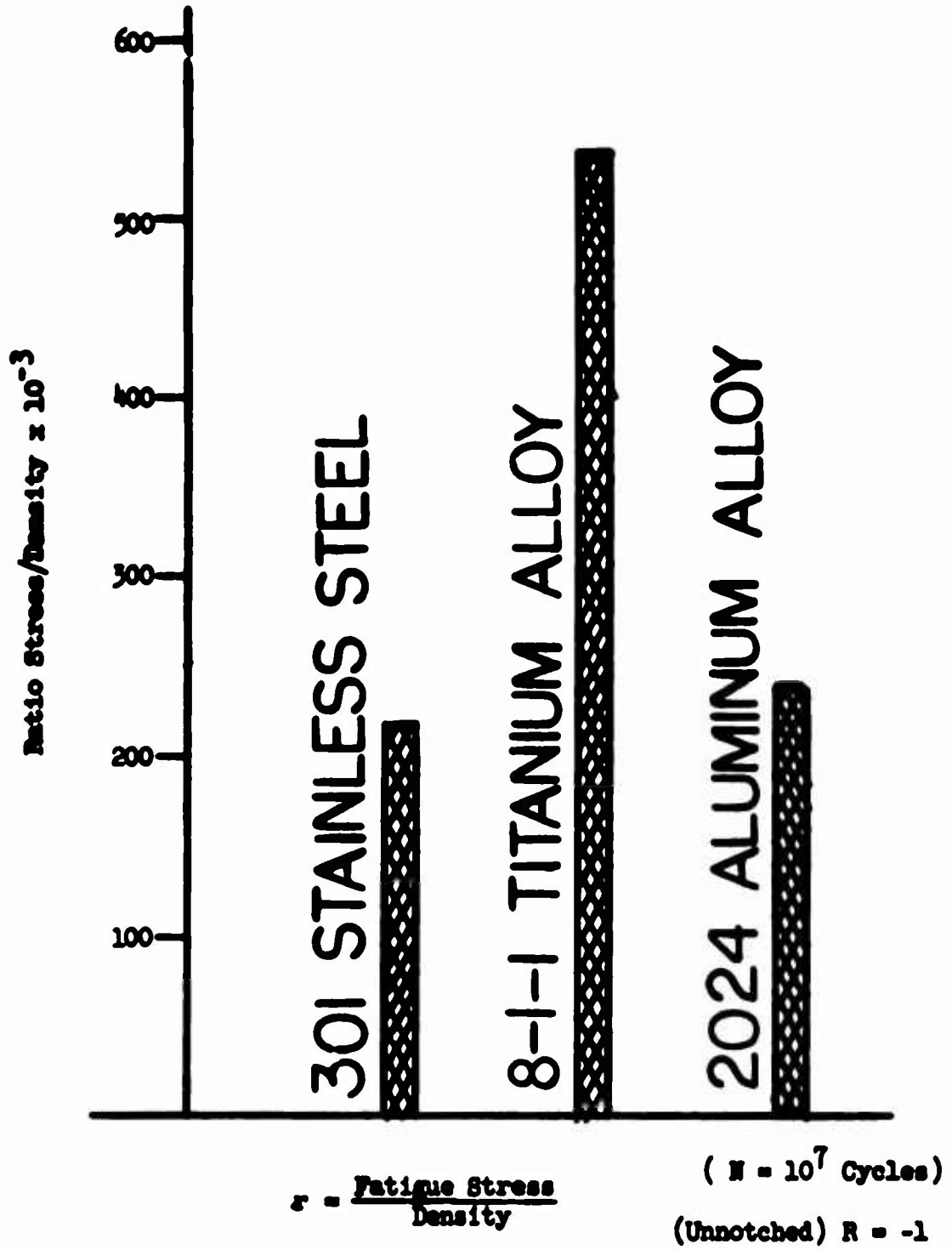
Figure 11. Fatigue Life/Density.

OIL OUTLET LINE
(14 PLACES)

FUEL SHUT OFF VALVE
(14 PLACES)

FUEL OUTLET LINE, UPPER
(14 PLACES)

FUEL INTERCONNECT VALVE

FUEL PRESSURE TRANSMITTER

ELECTRICAL HARNESS OUTLET
(14 PLACES)

ROTOR HUB ASSY (REF)

ROTOR MAST

ELECTRICAL HARNESS

AIR LINE

AIR INLET MANIFOLD

MANIFOLD ANTI-ROTATION
BRACKET

FUEL INLET MANIFOLD

OIL OUTLET MANIFOLD

FUEL OUTLET MANIFOLD, UPPER

FUEL OUTLET LINE, LOWER
(14 PLACES)

FUEL OUTLET MANIFOLD, LOWER

FUEL CHECK VALVE

AIR OUTLET LINE
(14 PLACES)

OIL LINE

SYSTEMS SUPPORT

FUEL LINE

AIR INLET LINE

FUEL INLET LINE

SLIP RING ASSY

OIL INLET LINE

FWD

UP

ROTOR
SHAFT

Figure 12. Transfer System Assembly - Main Rotor.

59

COLLAR-ISOLATION ARM
(MAST MOUNTED)

DREM-20-050 (SOUTHWEST)
ROD END
TYP 24 PLACES

MS20201KP20A
BALL BEARING
4 PLACES

RS 32 (MC GILL)
NEEDLE BEARING (2)
4 PLACES

ISOLATION ARM-
HUB TEETER (4)

52.00

WL
416.50

AXIS OF
ROTATION

WALKING BEAM (4)
(PITCH INPUT)

ROTOR BLADE (REF)

MS20202KP49B
BALL BEARING (4)

KG350AR (KAYDON)
BALL BEARING
DUPLEX PAIR

DRIV

SWA

SWA

SWASH

SECTION A-
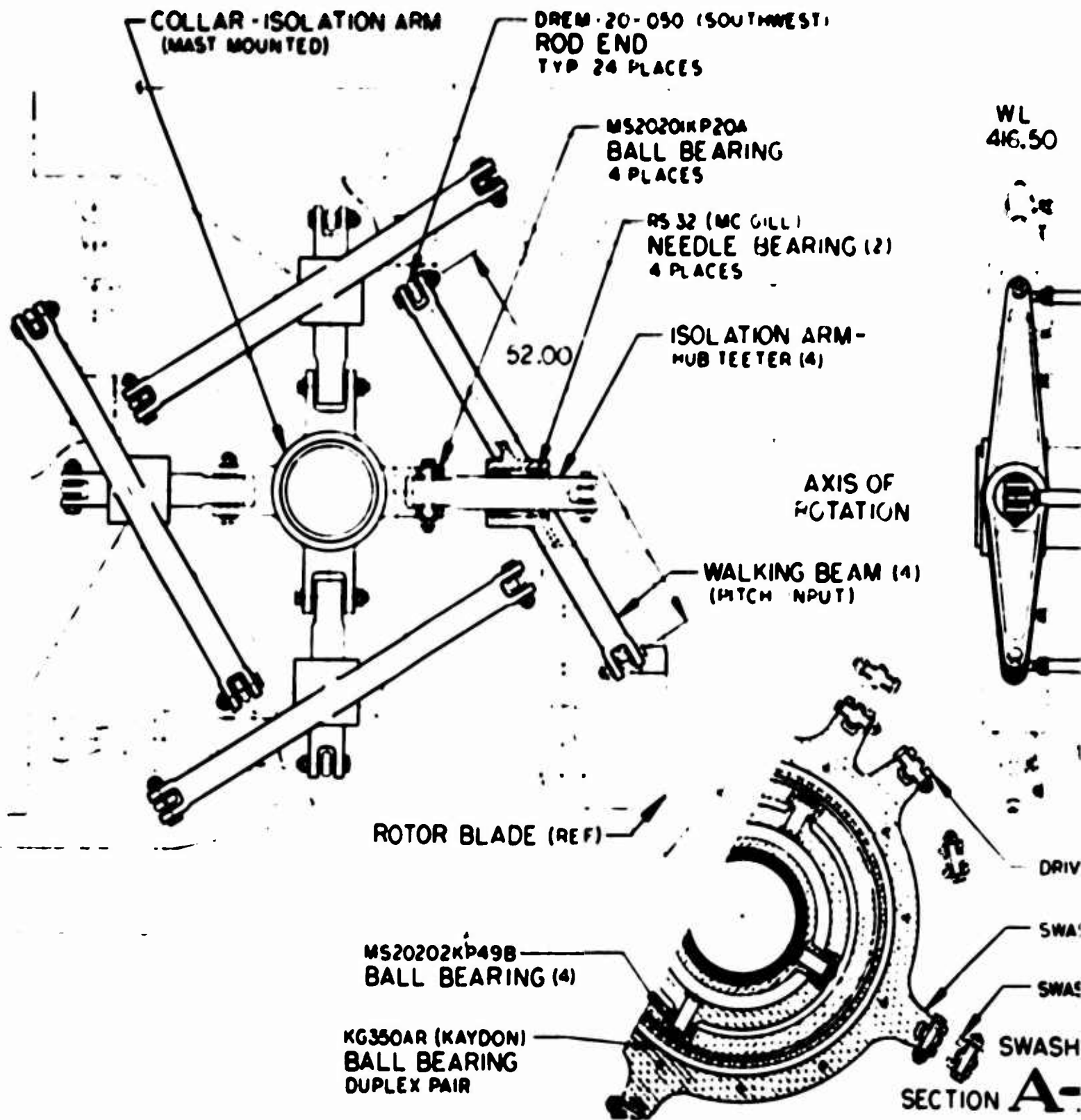
Figure 13. Flight Control
Main Rotor.

61

A

ROTOR HUB (REF)

DRIVE SCISSORS

BLFR-8-045 (SOUTHWEST)
SPHERICAL BALL BEARING
2 PLACES

POWER CYLINDER (3)
(TANDEM HYDRAULIC)

A

SWASH PLATE

LUG-ISOLATION LINK
(HUB MOUNTED)

HOLDING SCISSORS

MAST (REF)

LUG-PITCH INPUT

SLEEVE
(STATIONARY)

COLLAR-SLIDE
(TEFLON LINED)

A

LUG

WL
338.75

UP

ROTOR BLADE (REF)
(AT MID PITCH)

FORWARD

H PLATE OUTPUT (4)

H PLATE INPUT (3)

PLATE

ROTATING ← → NON-ROTATING

A

Installation -

B

PRIMARY SYSTEM

SECONDARY SYSTEM

PRESSURE
RETURN
SUCTION

COLLECTIV
LONGITUDI
CYCLIC

Figure 14. Controls - Hyd

PRESSURE

RETURN

SUCTION

PILOT INPUT

COLLECTIVE &
LONGITUDINAL
CYCLIC

COLLECTIVE &
LONGITUDINAL
CYCLIC

COLLECTIVE &
LATERAL
CYCLIC
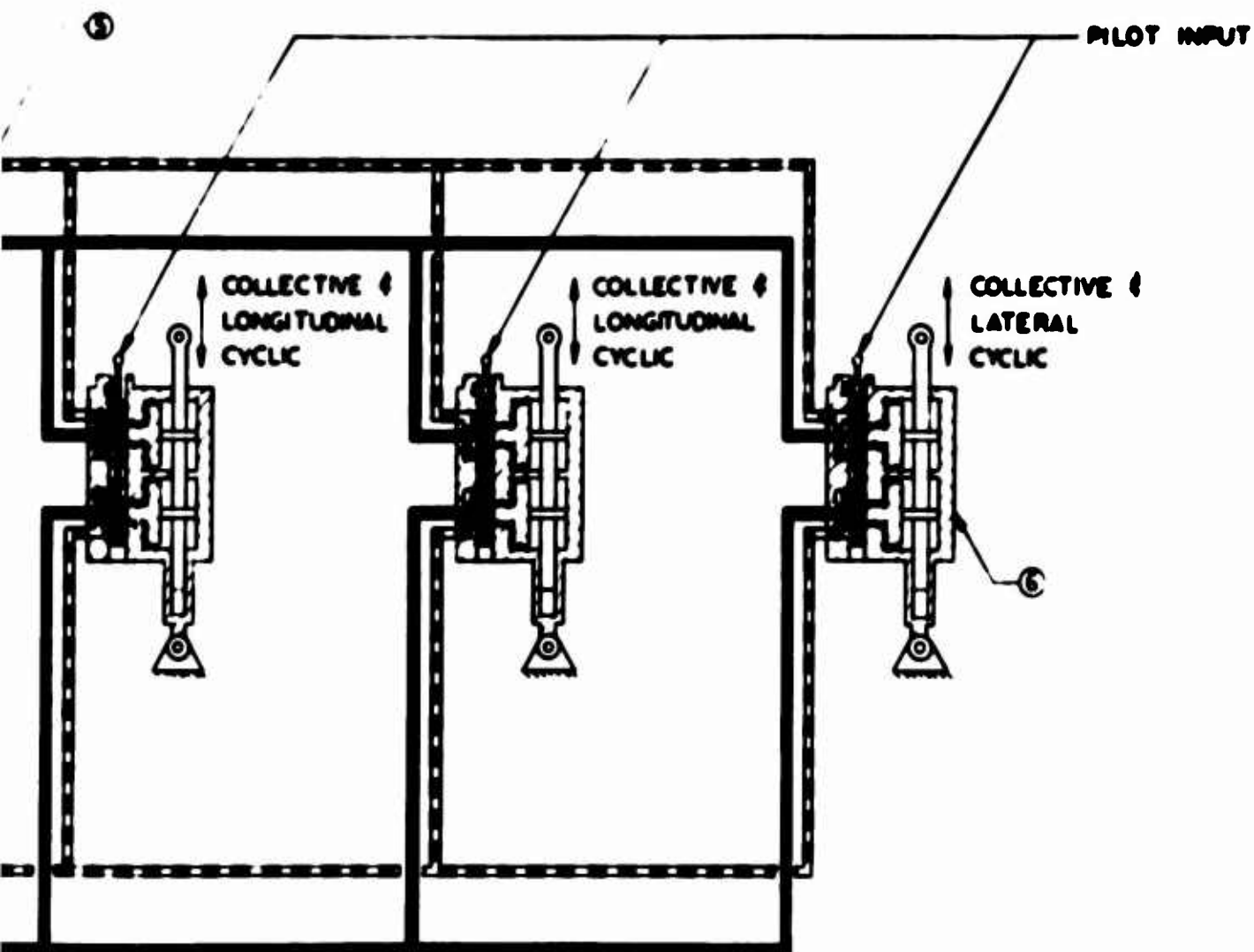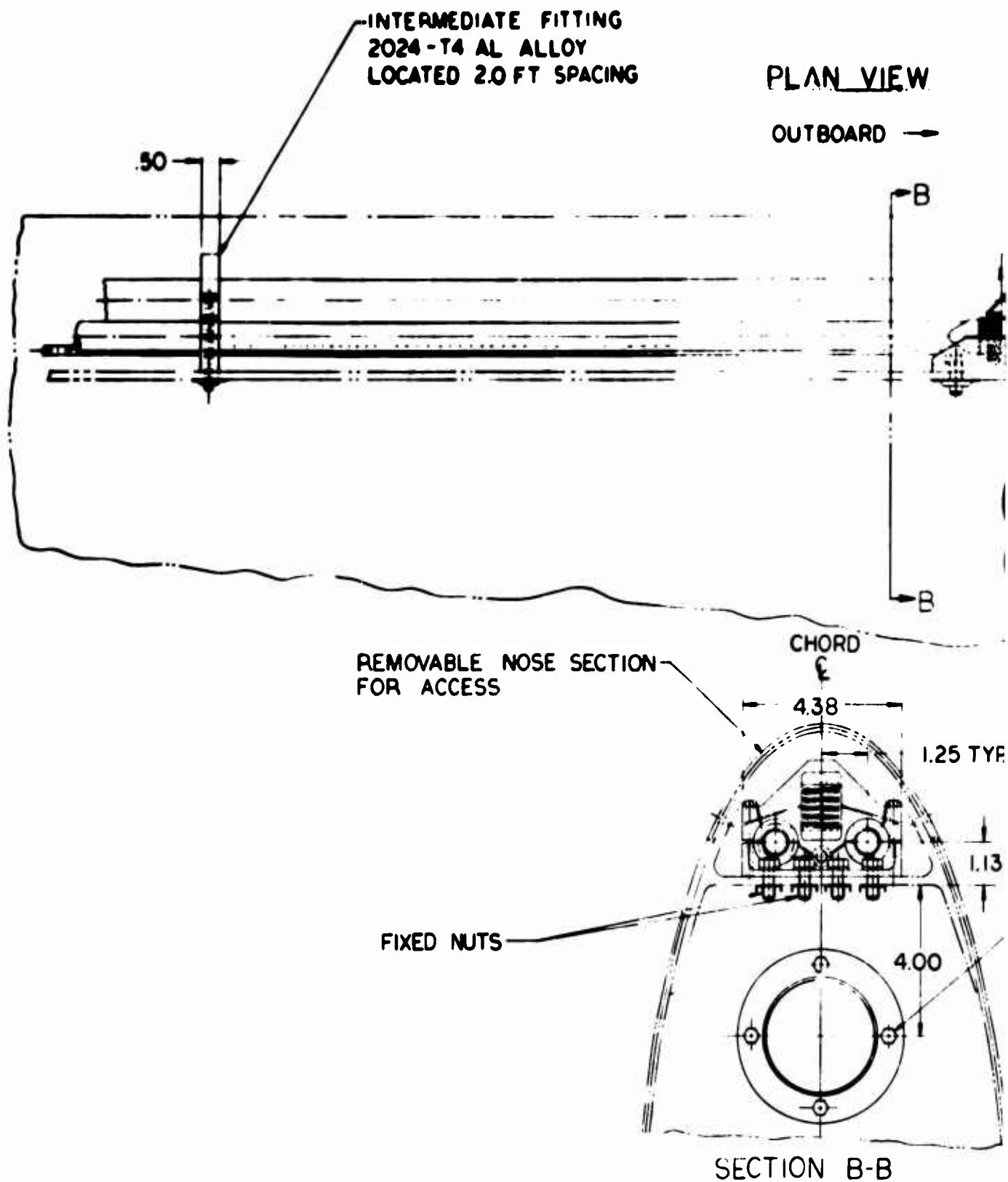
ure 14. Controls - Hydraulics System Schematic.

B

INTERMEDIATE FITTING
2024-T4 AL ALLOY
LOCATED 2.0 FT SPACING

PLAN VIEW

OUTBOARD →

.50

B

B

REMOVABLE NOSE SECTION
FOR ACCESS

CHORD
℄

4.38

1.25 TYP

1.13

FIXED NUTS

4.00

SECTION B-B

Figure 15.   Engine Systems Retention - Rotor Blade.

2024-T4 AL. ALLOY
END FITTING
TYPICAL BOTH ENDS

RIBBON WIRING
INTER-LAYER BONDED
-ELECTRICAL BUNDLE

ENGINE

OIL REPLENISHING LINE
.035 X ¼ DIA. 5052-SO
AL. ALLOY

BRAZED
PERMANENT COUPLINGS

VERTICAL FIREWALL

OIL REPLENISHING
VALVE

AIR START
LINE, 3 INCH DIA.
.020 TITANIUM
UNALLOYED
(ASTM-GRADE 3)

BULKHEAD FITTING

FUEL LINES
.058 X ¾ DIAMETER
TITANIUM UNALLOYED
(ASTM-GRADE 3)

STA.
636.00

WELDED FORK JOINT

1.25 TYP.

CHORD

ELECTRICAL CONNECTOR

OIL REPLENISHING LINES

2.06

.65
TYP.

.92

FUEL LINES

2.40

1.13

TUBE FLANGE
ATTACHMENT

.00

AIR START LINE

SECTION A-A

Blade.

B

Figure 16. Engine Instrument System - Block Diagram.

67

Figure 17. Instrument Arrangement - Cockpit.

68

MAIN ENGINE MOUNT
AFT ENGINE MOUNT
OIL LINES
OIL COOLER

OIL REPLENISHING LINE
FUEL LINES
ELECTRICAL BUNDLE
AIR START LINE

FIRE WALL
CONTINENTAL AVIATION & ENGINEERING MODEL 357-1 ENGINE
FUEL PUMP
FUEL CONTROL ACTUATOR
FUEL CONTROL
ENGINE NACELLE
HORIZONTAL FIREWALL

ELECTRICAL DISCONNECT
ROTOR BLADE FIREWALL
OIL SHUT OFF VALVES

Figure 16. Power Plant and Systems Installation (Isometric).

ENGINE AIR INLET
(UPPER)

INLET CENTER
BODY-(UPPER)

AIR START CONNECTION-
AIRCRAFT SUPPLY

FUEL PUMP

FUEL CONTROL
ACTUATOR

FUEL CONTROL

INLET CENTER
BODY-(LOWER)

ENGINE AIR INLET
(LOWER)

110H-6.5
MAIN ENGINE M

FWD

STA
100.00
NAC

AIR START CONNECTION
GROUND SUPPLY

FUEL
DRAIN

VIEW LOOKING OUTBOARD

71

Figure 19. Engine Installation

A

−1108-6.30
MAIN ENGINE MOUNT

1108-650
NACELLE INSTALLATION

ENGINE
(UPPER)

−1108-530
MAIN ROTOR
BLADE (REF)

13.50

13.50

−1108-631
REAR ENGINE MOUNT

ENGINE
(LOWER)

CONTINENTAL ENGINE
MODEL 357-1

FUEL
DRAIN
JTBOARD
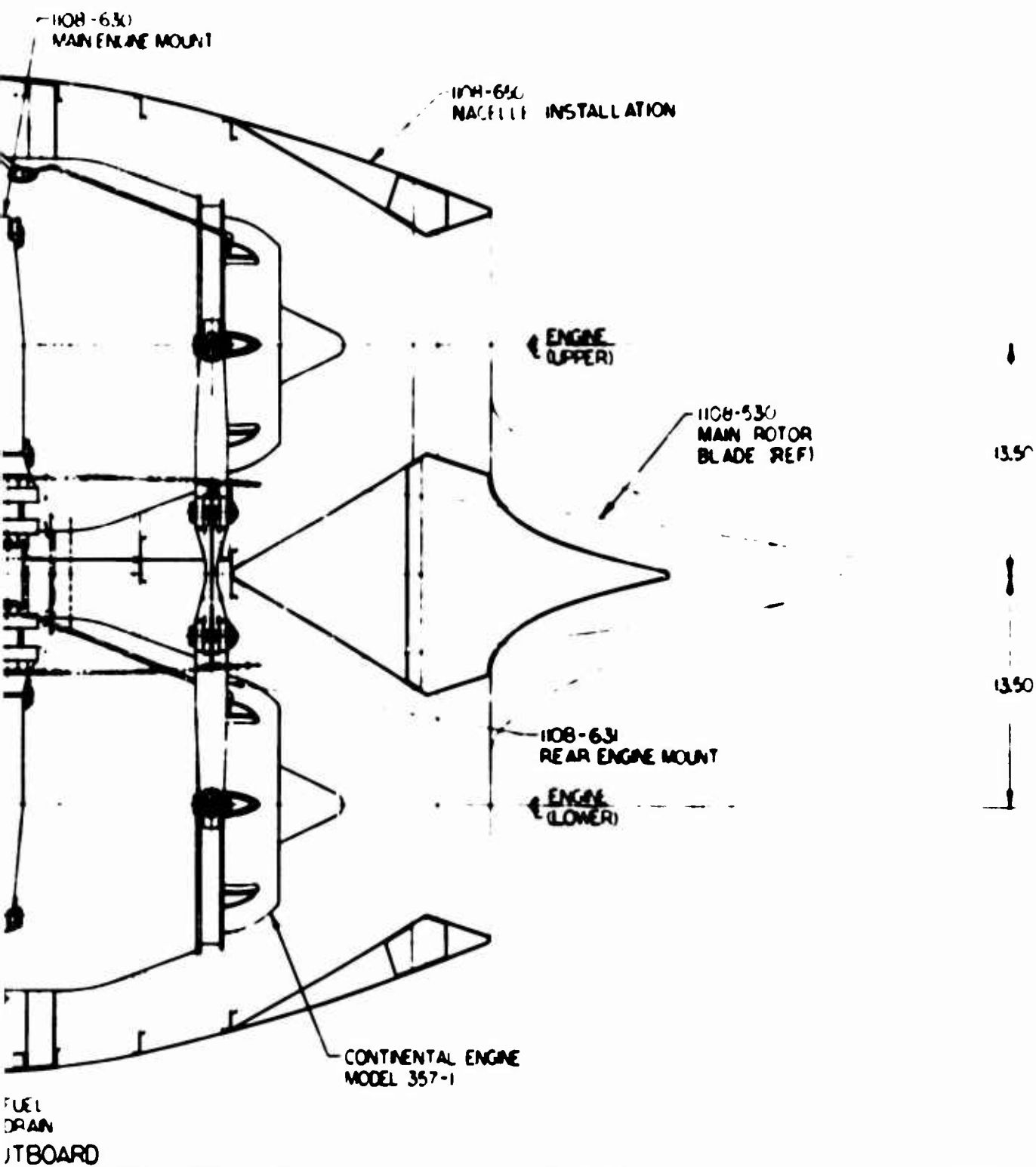
ne Installation - Inboard Profile.

B

ROTOR BLADE
STA (REF)
67200

CONTINENTAL AVIATION
ENGINEERING MODEL 357-1
ENGINE (REF) TYP

3.25

6.25
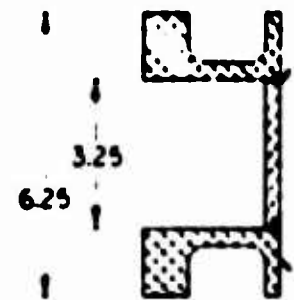
SECTION A-A
FULL SCALE

TITANIUM
ALLOY
6AL-4

BLADE
STA.
64-

27.00 ROTOR BLADE
CHORD LINE

24.00

13.00
TYP

500

30°
TYP

C   C

B   B

BLADE
STRUCTUR

NACELLE CONTOUR
AT STA 118.50
1108 650 (REF)

2.38

SECTION D-D
FULL SCALE

L25 DIA
TYPE N
TITANIU
7AL-122

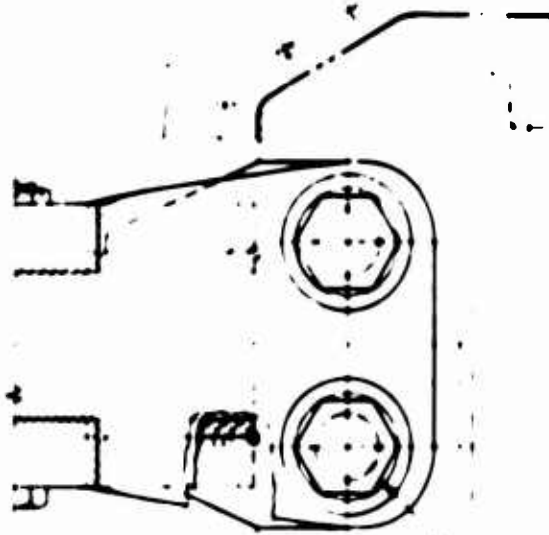OUTBD
VIEW LOOKING AFT

Figure 20. Main Engine Mou

A

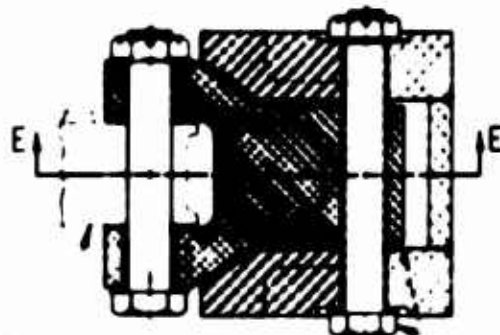ELECTRON BEAM
WELDING

NACELLE
STA
48.50

TITANIUM
ALLOY
6AL-4

VIEW B-B
(BLADE ATTACHMENT)
FULL SCALE

4.5

2.25

BLADE TIP STRUCTURE
1K-B 552 REF

V BOLT

1.56 DIA BOLT
TITANIUM ALLOY
7AL-12ZR

BLADE TIP
STRUCTURE

E —— E

SECTION C-C
(EXPANSION LINK)
FULL SCALE

ENGINE
LUG REF

LINK
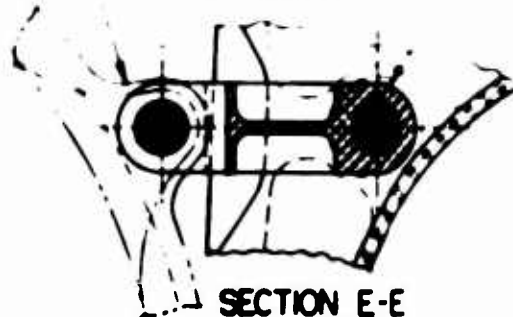
1.25 DIA BOLT
TYPE NAS 464
TITANIUM ALLOY
7AL-12ZR

FWD

1.25 DIA BOLT
TYPE NAS 464
TITANIUM ALLOY
7AL-12ZR

SECTION E-E
FULL SCALE

Engine Mount - Tip Turbojet.

B

NAC STA
133 52

ROTO
STA

BLADE TIP STRUCTURE
1108-532 (REF)

B

B

27.00

17.50

FWD

7.25

MOUNT

9.8(

.625 DIA BOLTS
TYPE NAS 464
TITANIUM ALLOY
7AL-12ZR (TYP)

1.750

1.00

SPHERICAL
BEARING

SECTION A-A
FULL SCALE
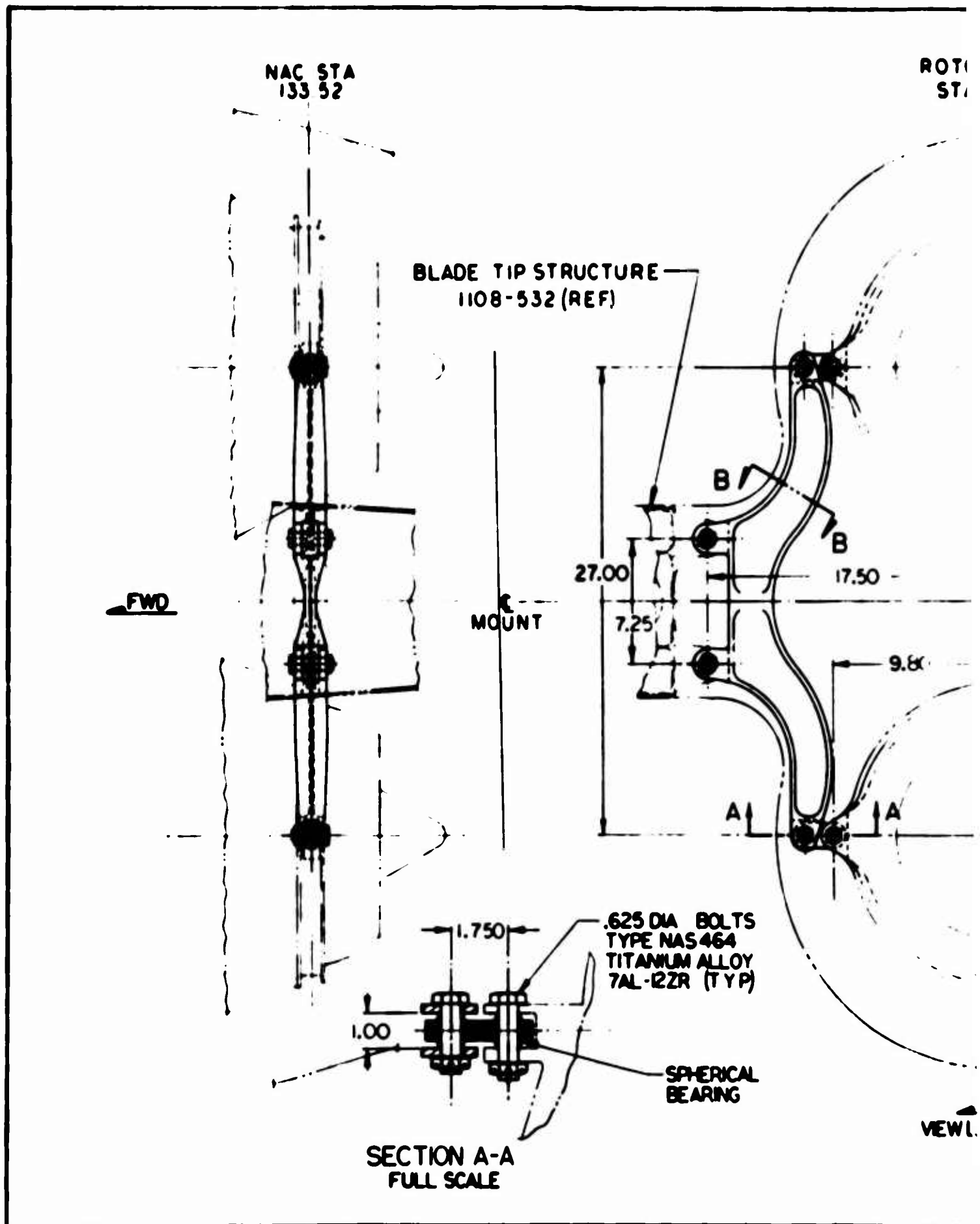
VEW L.

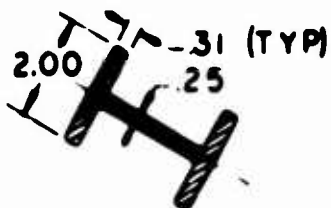Figure 21. Aft Engine M

A

ROTOR BLADE
STA 672.00

CONTINENTAL AVIATION & ENGINEERING
MODEL 357-I ENGINE (REF)

2.00   .31 (TYP)
        .25

SECTION B-B
FULL SCALE
TITANIUM ALLOY FORGING
6AL-4V

B

17.50

ROTOR BLADE
CHORD LINE

9.80

NACELLE CONTOUR
AT STA 134.72N
1108-650 (REF)

A

CONTINENTAL AVIATION & ENGINEERING
MODEL 357-I ENGINE (REF)
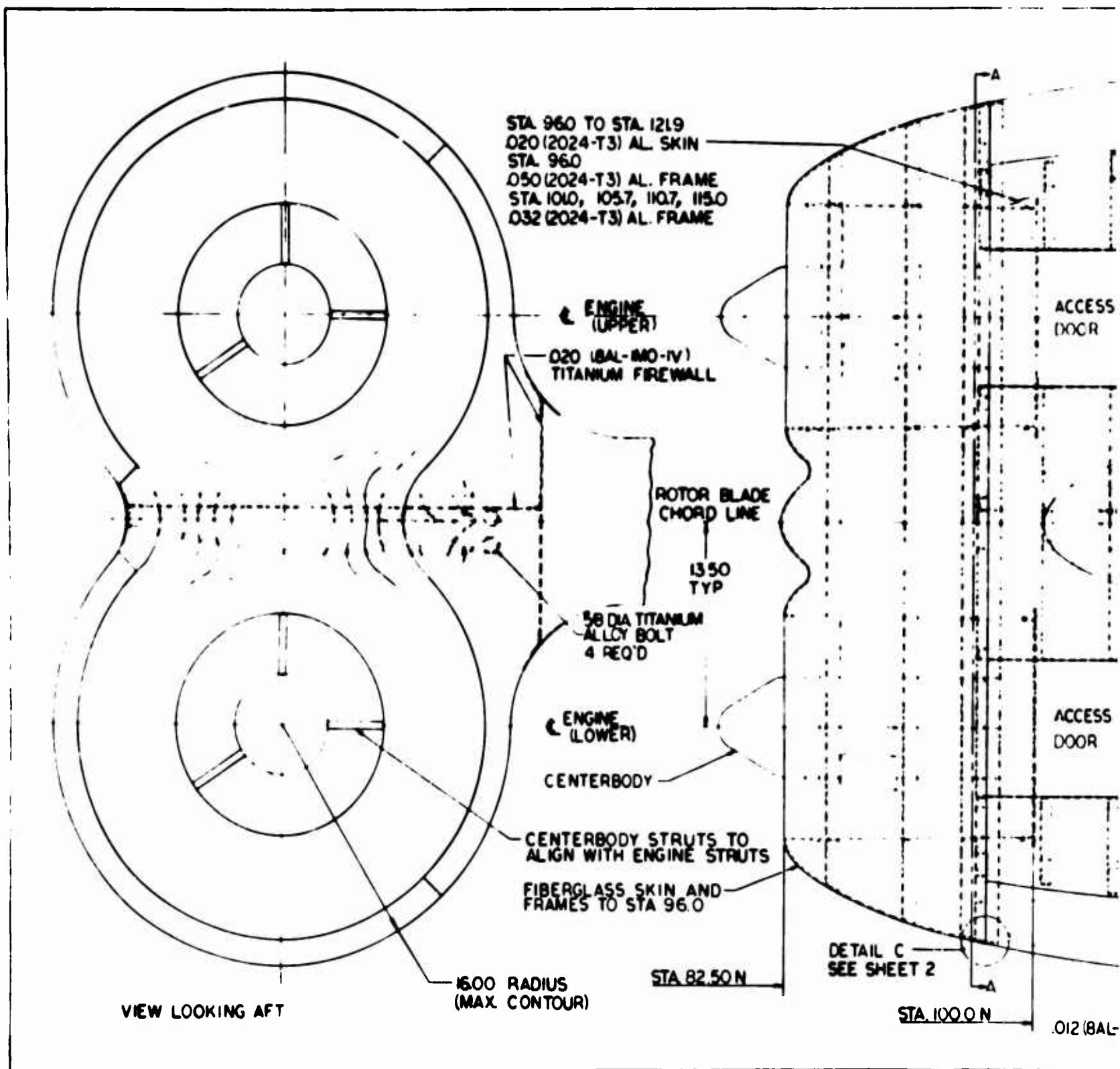
INBD

VIEW LOOKING FWD

Aft Engine Mount - Tip Turbojet.

B

STA. 96.0 TO STA. 121.9
.020 (2024-T3) AL. SKIN
STA. 96.0
.050 (2024-T3) AL. FRAME
STA. 101.0, 105.7, 110.7, 115.0
.032 (2024-T3) AL. FRAME

¢ ENGINE
(UPPER)

.020 (8AL-1MO-1V)
TITANIUM FIREWALL

ROTOR BLADE
CHORD LINE

13.50
TYP

5/8 DIA TITANIUM
ALLOY BOLT
4 REQ'D

¢ ENGINE
(LOWER)

CENTERBODY

CENTERBODY STRUTS TO
ALIGN WITH ENGINE STRUTS

FIBERGLASS SKIN AND
FRAMES TO STA 96.0

16.00 RADIUS
(MAX. CONTOUR)

VIEW LOOKING AFT

ACCESS
DOOR

ACCESS
DOOR

STA. 82.50 N

DETAIL C
SEE SHEET 2

STA. 100.0 N

.012 (8AL-

Figure 22. Nacelle Ins

77

A

—D ⌐B

STA. 122.7N

CONTINENTAL AVIATION
ENGINEERING MODEL 3571
ENGINE (REF)

ACCESS
DOOR

ACCESS
DOOR

STA. 150.84N

STA. 121.9 AFT
TITANIUM ALLOY
.025 FRAMES (6AL-4V)
.025 SKINS (8AL-IMO-IV)

STA 115N—E

.012 (8AL-IMO-IV) TITANIUM FIREWALL

COOLING AIR INLET
10 IN DIA HOLES
30 REQ'D PER PANEL

VIEW LOOKING OUTBOARD

celle Installation - Tip Turbojet, Sheet 1.

B

SECTION D-D
COWL HINGE

QUARTER TURN
FASTENER
MIL-F-5591

STA 96.0 N
REF

DETAIL C
FULL SIZE

FIREWALL

NOSE FAIRING
ATTACH FITTING FOR
MS 20005 BOLT

FORWARD ACCESS DOORS
ALUM. HONEYCOMB

OUTB'D

SECTION A-A

Figure 23. Nacelle Installation -

A

REMOVABLE
PANEL

FIREWALL

TITANIUM ALLOY
PANEL ATTACH
BOLTS
2 REQ'D PER
PANEL

'URN

REMOVABLE
PANEL

SECTION B-B

ation - Tip Turbojet, Sheet 2.

B

Figure 24. Power Management System - Block Diagram.
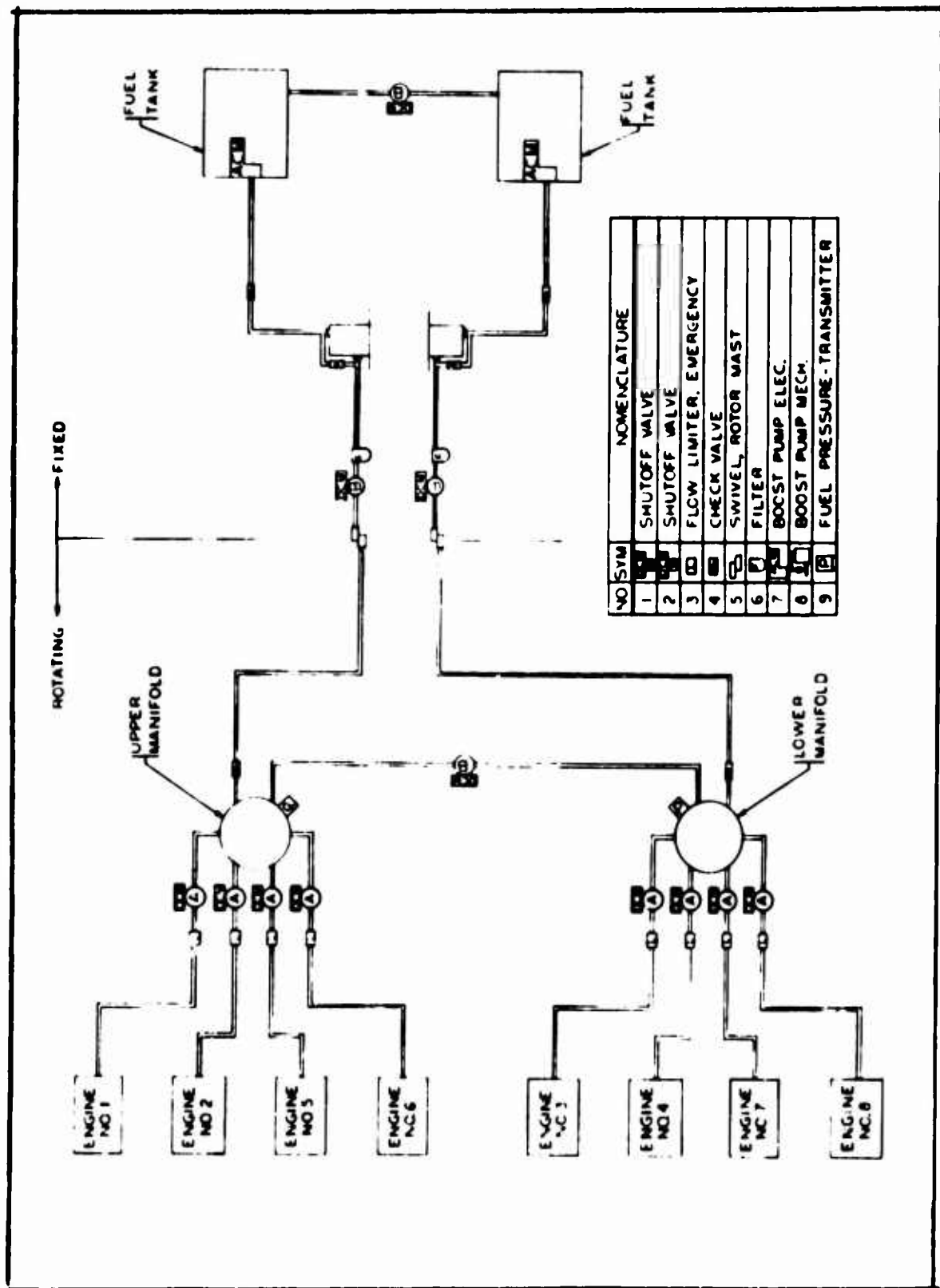
Figure 25. Fuel System - Schematic Diagram.

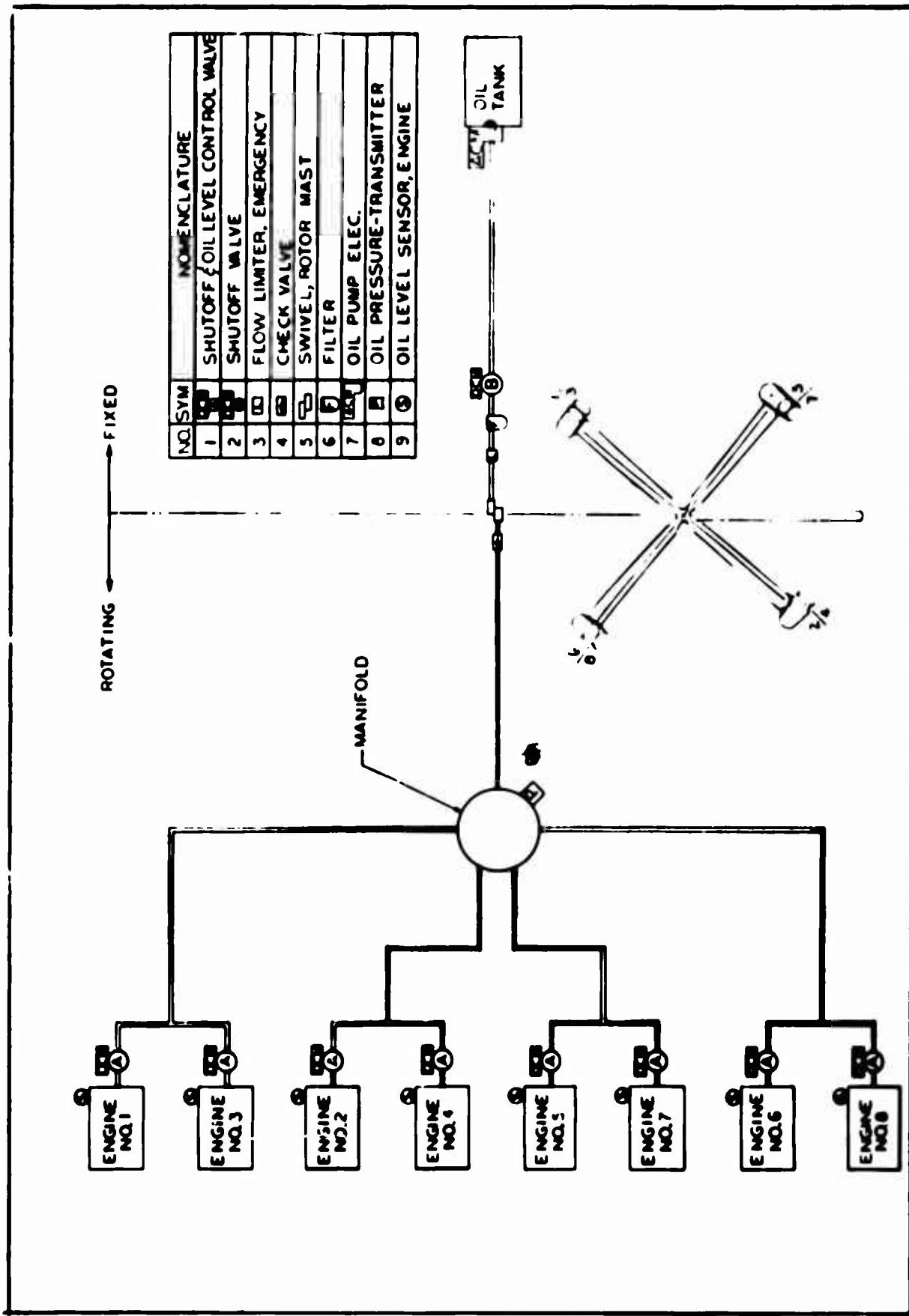| NO	SYM	NOMENCLATURE
1		SHUTOFF VALVE
2		SHUTOFF VALVE
3		FLOW LIMITER, EMERGENCY
4		CHECK VALVE
5		SWIVEL, ROTOR MAST
6		FILTER
7		BOOST PUMP ELEC.
8		BOOST PUMP MECH.
9		FUEL PRESSURE - TRANSMITTER

Figure 26. Oil Replenishment System - Schematic Diagram.

NO	SYM	NOMENCLATURE
1		SHUTOFF {OIL LEVEL CONTROL VALVE
2		SHUTOFF VALVE
3		FLOW LIMITER, EMERGENCY
4		CHECK VALVE
5		SWIVEL, ROTOR MAST
6		FILTER
7		OIL PUMP ELEC.
8		OIL PRESSURE-TRANSMITTER
9		OIL LEVEL SENSOR, ENGINE

FIXED

ROTATING

OIL TANK

MANIFOLD

ENGINE NO.1
ENGINE NO.3
ENGINE NO.2
ENGINE NO.4
ENGINE NO.5
ENGINE NO.7
ENGINE NO.6
ENGINE NO.8
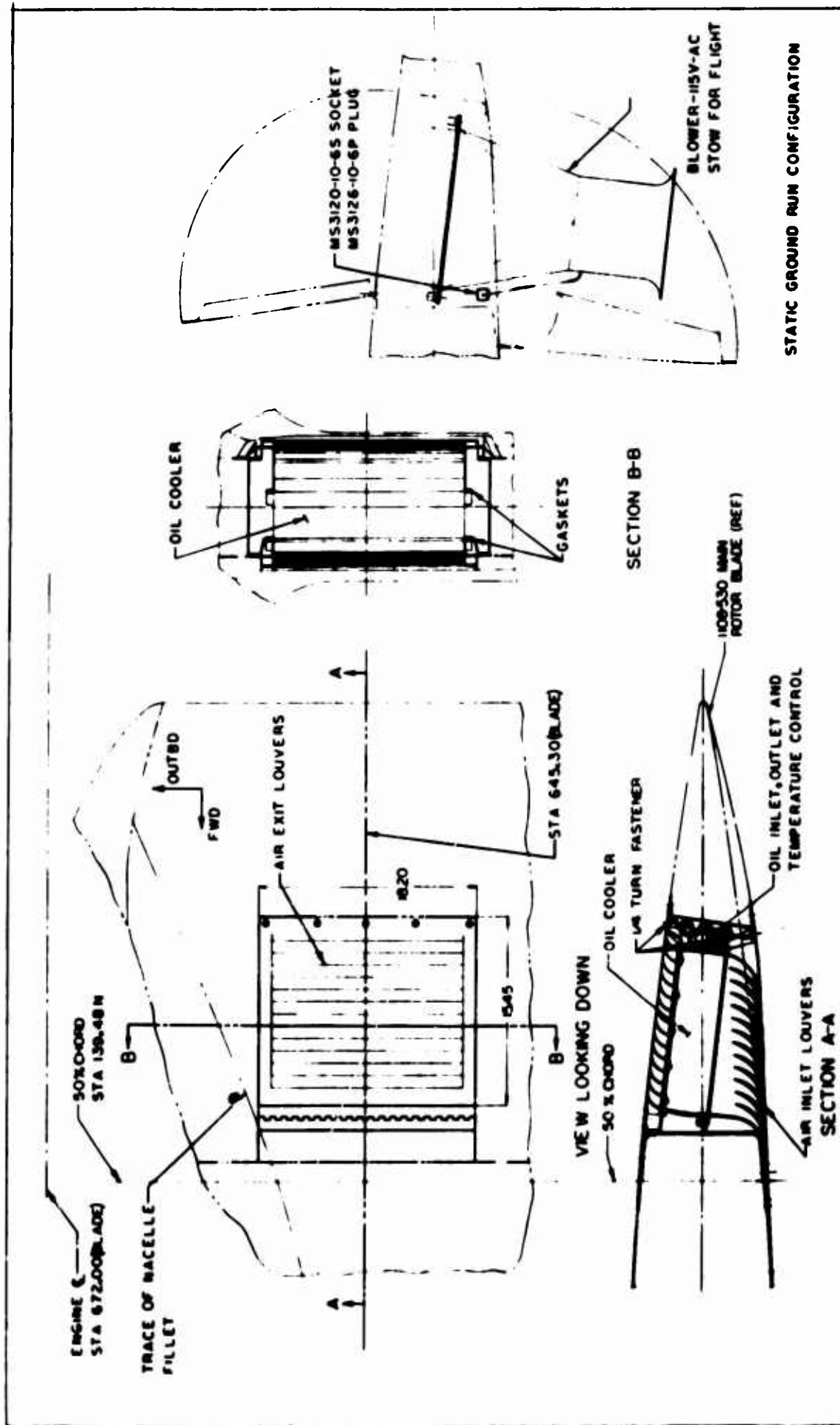
Figure 27. Oil Cooler Installation - Tip Turbojet.

ENGINE ₵
STA 672.00(BLADE)

TRACE OF NACELLE FILLET

50% CHORD
STA 138.48 IN

OUTBD

FWD

AIR EXIT LOUVERS

16.20

15.45

STA 645.30(BLADE)

B

B

A

A

VIEW LOOKING DOWN

50% CHORD

OIL COOLER

1/4 TURN FASTENER

1108-530 WMH
ROTOR BLADE (REF)

OIL INLET, OUTLET AND
TEMPERATURE CONTROL

AIR INLET LOUVERS

SECTION A-A

OIL COOLER

GASKETS

SECTION B-B

MS3120-10-6S SOCKET
MS3126-10-6P PLUG

BLOWER-115V-AC
STOW FOR FLIGHT
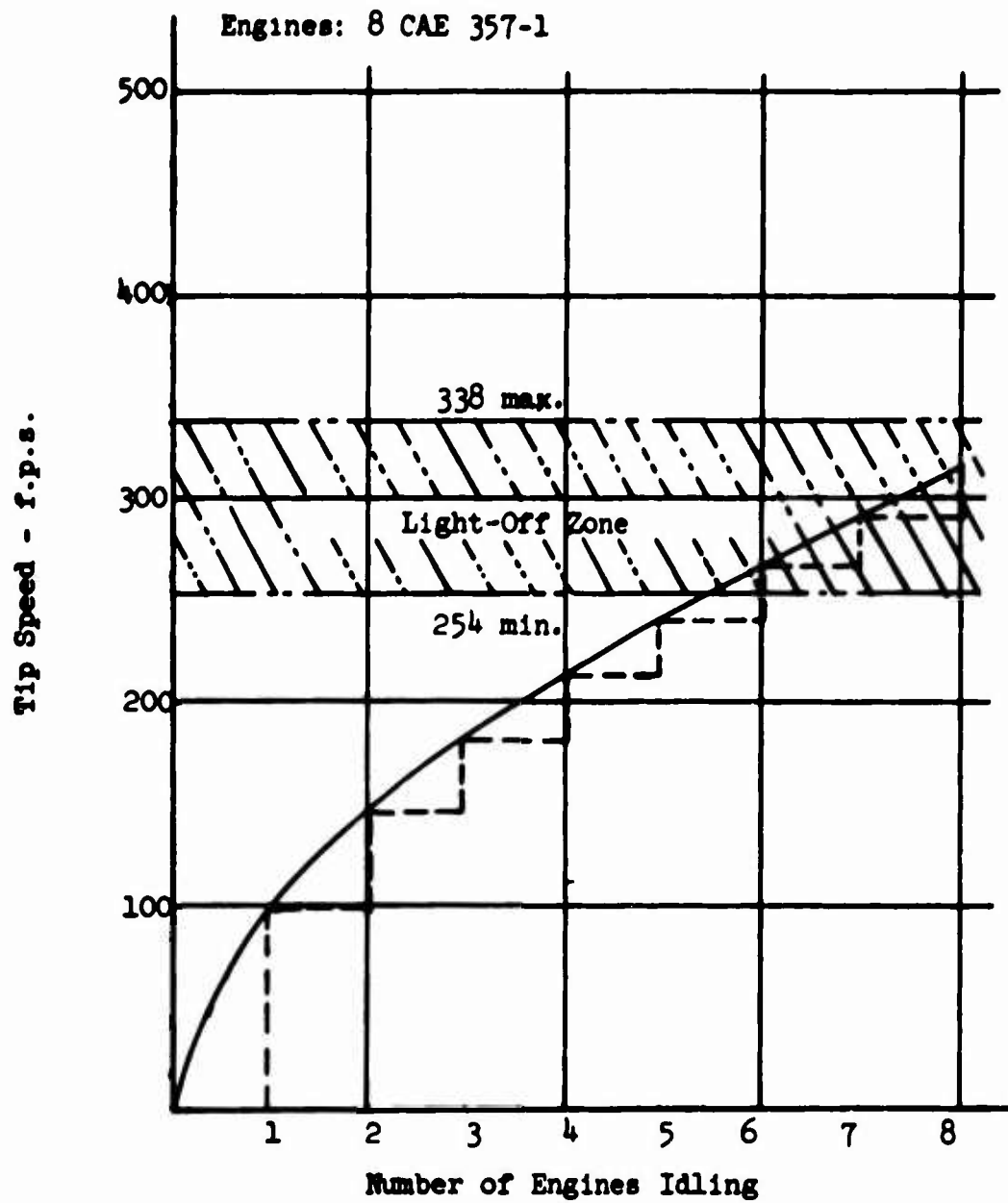
STATIC GROUND RUN CONFIGURATION
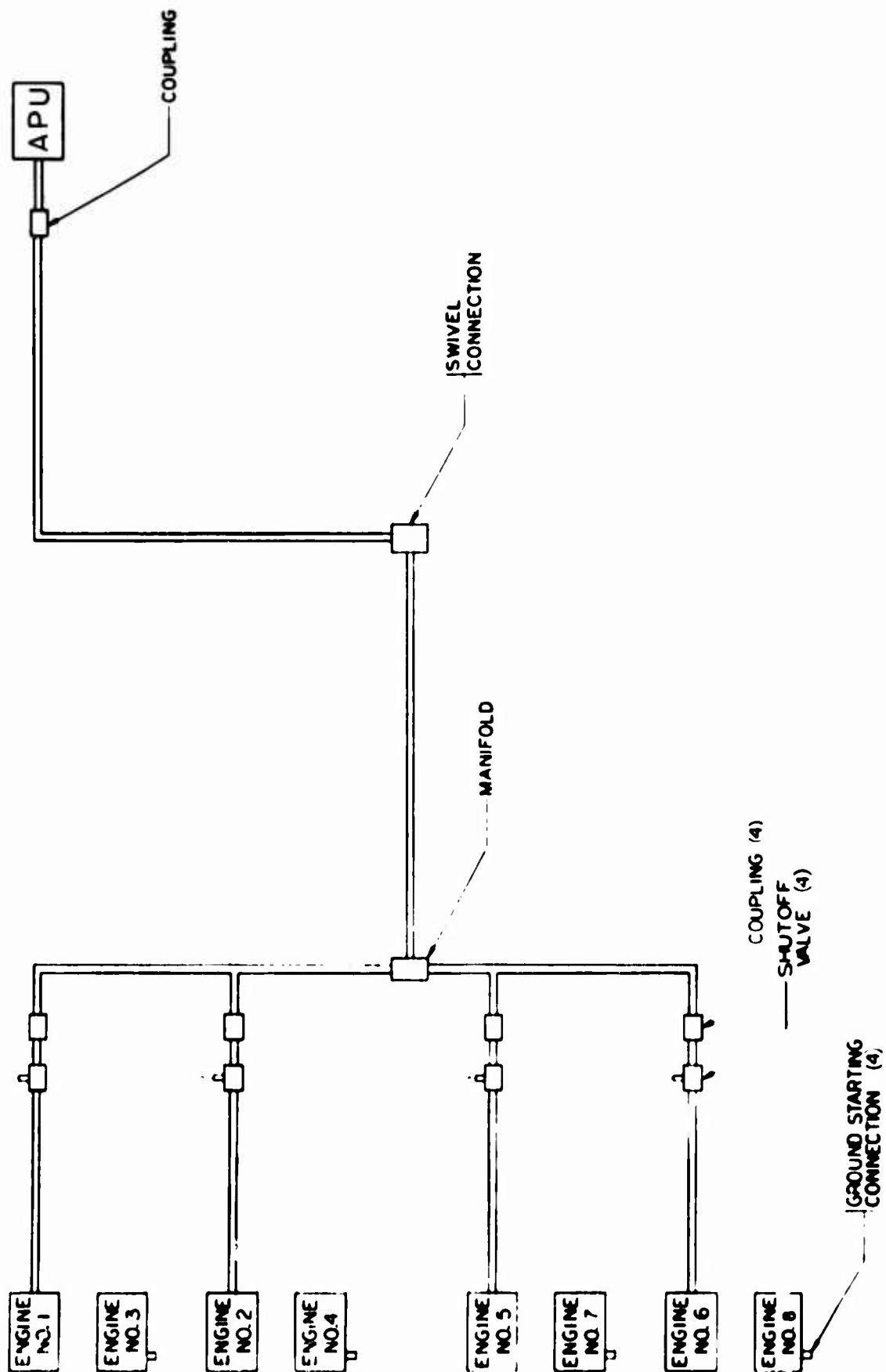
84

Figure 28. Tip Speed Versus Number of Engines Idling.

Figure 29. Air Impingement Starting System - Block Diagram.

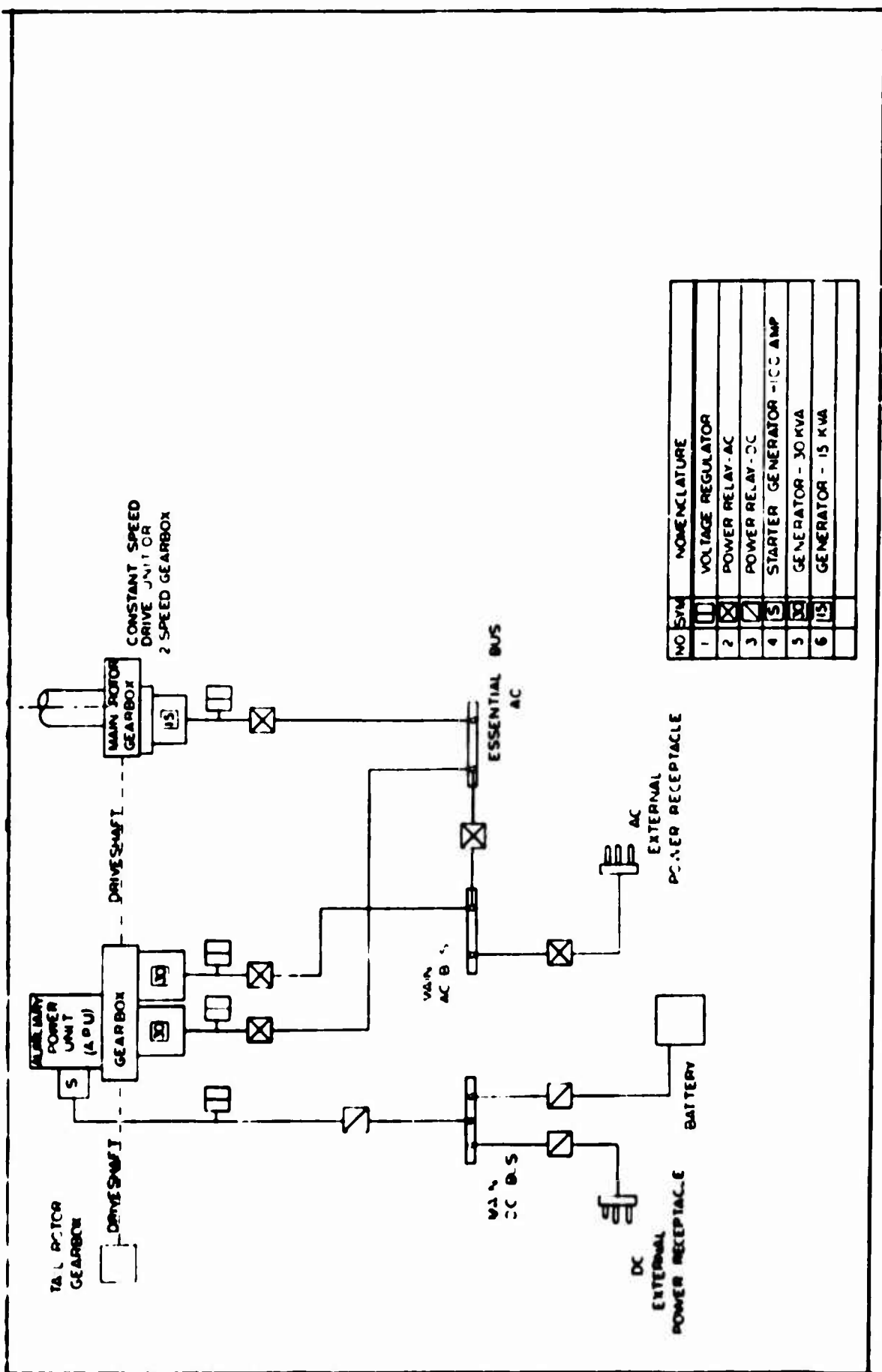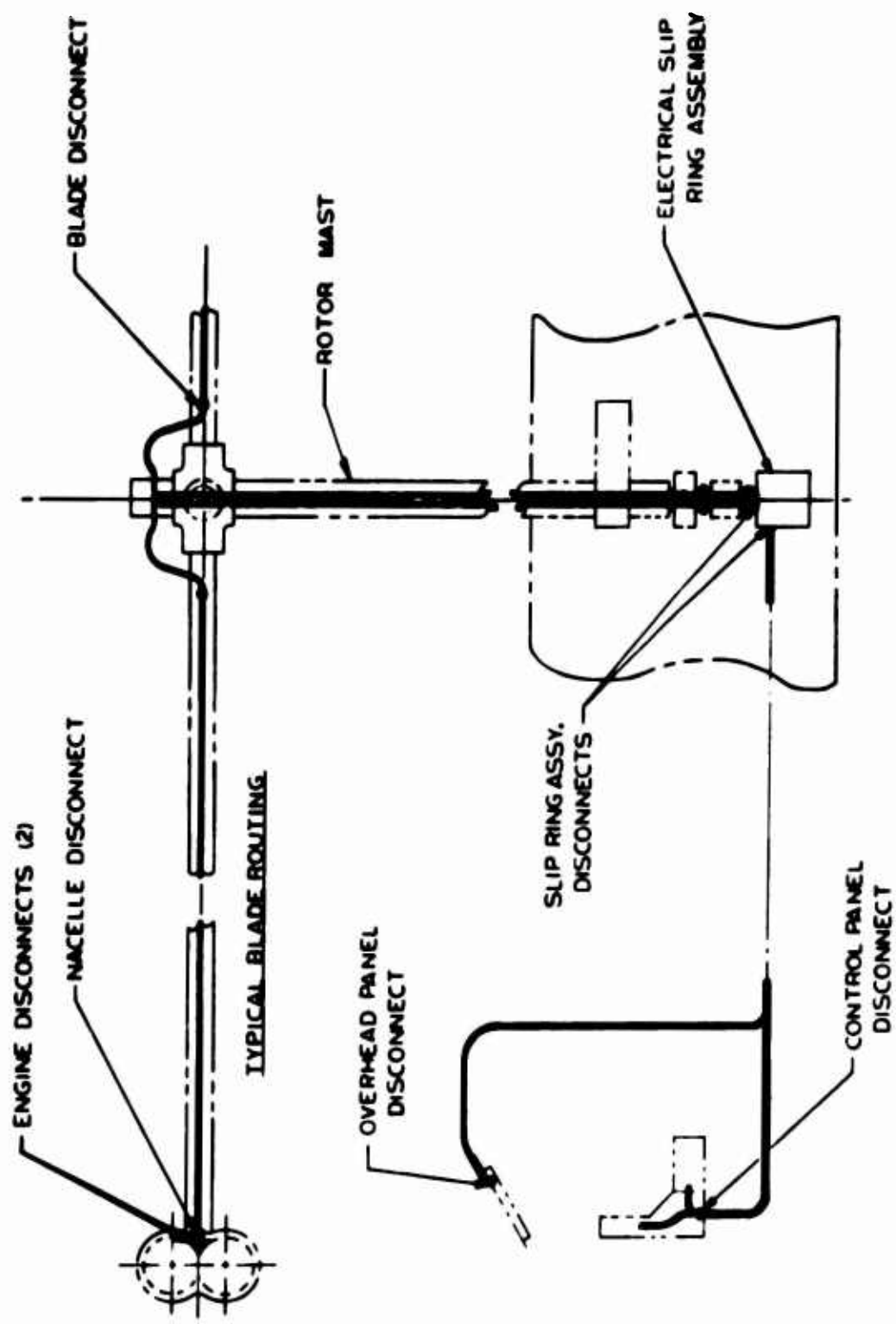**Figure 30. Electrical Power Distribution System - Block Diagram.**

NO	SYM	NOMENCLATURE
1		VOLTAGE REGULATOR
2		POWER RELAY-AC
3		POWER RELAY-DC
4	S	STARTER GENERATOR - 100 AMP
5		GENERATOR - 30 KVA
6	15	GENERATOR - 15 KVA

BLADE DISCONNECT

ROTOR MAST

ELECTRICAL SLIP RING ASSEMBLY

ENGINE DISCONNECTS (2)

NACELLE DISCONNECT

TYPICAL BLADE ROUTING

SLIP RING ASSY. DISCONNECTS

OVERHEAD PANEL DISCONNECT

CONTROL PANEL DISCONNECT

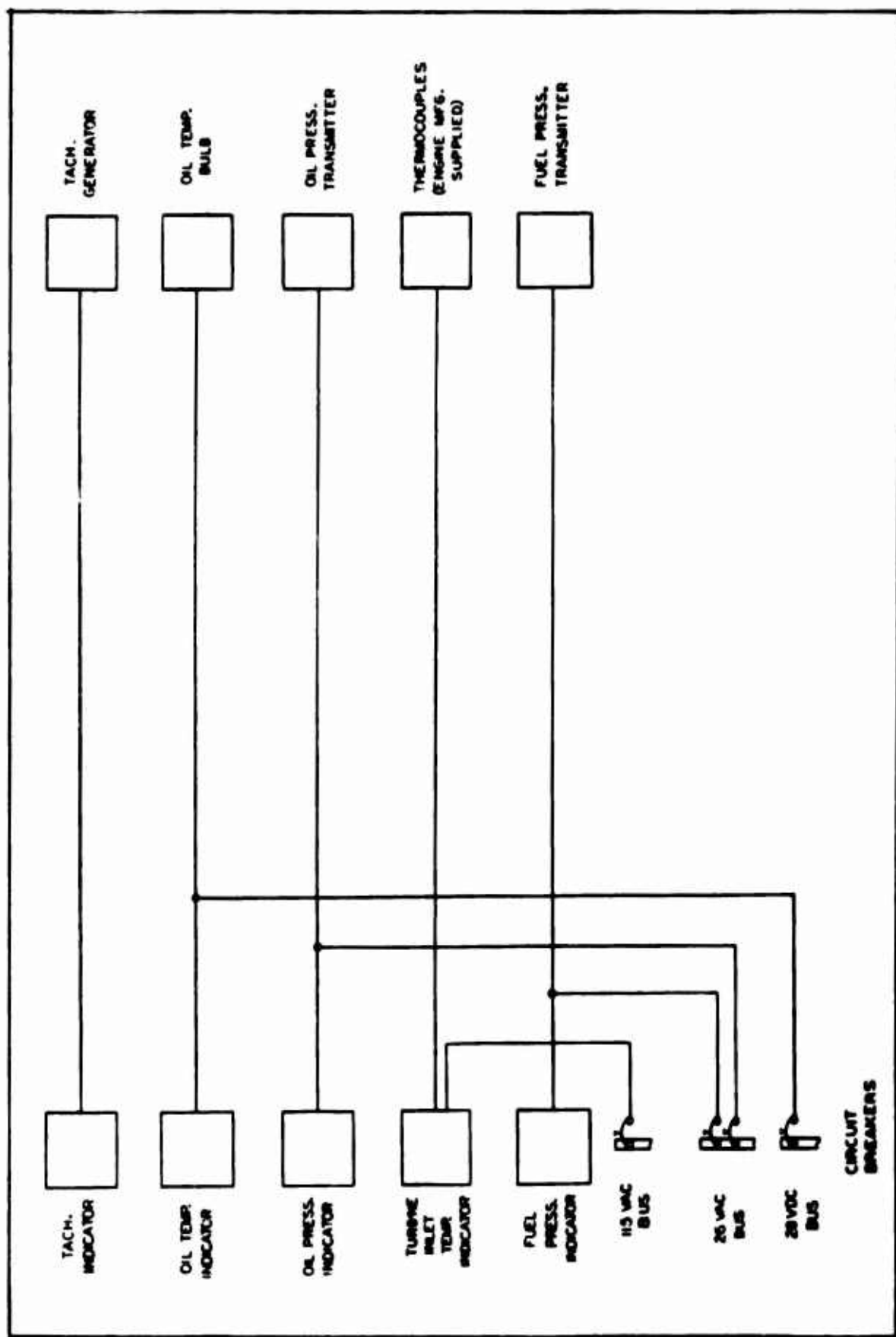Figure 31. Electrical System Disconnect and Harness Routing.

Figure 32. APU Instrument System - Block Diagram.

89

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5.0 APPENDIX I

ENGINE POWER MANAGEMENT SYSTEM SPECIFICATION

5.1 **Scope** - This specification establishes the design and operating requirements for the power management and helicopter rotor speed control system for a helicopter powered by turbojet engines mounted at the rotor tips.

5.2 **Applicable Documents** - None

5.3 **Description** -

5.3.1 **Rotor and Engine Configuration** - The rotor and engine configuration to which this specification applies consists of a four-bladed rotor with two engines mounted at the tip of each blade. The engines are mounted in an over-under configuration with the engine centerlines essentially equally spaced above and below the blade centerline.

5.3.2 Operation Description -

5.3.2.1 Starting - All throttles initially should be in the "stop" position. Place the individual mode selector switches in the "auto" position, the master mode selector switch in the "manual" position, and the rotor speed selector to the lowest possible governed speed.

5.3.2.1.1 Odd Numbered Engines - Beginning with the lowest numbered engine and proceeding in sequence to the highest numbered engine, of the odd numbered engines, place the start switch in the "Ignite-Crank" position until the engine speed comes up to starting speed. Then move the throttle to the "Idle-Start" position, still holding the starting switch in the "Ignite-Crank" position until idle speed is achieved. Should an engine fail to start, proceed with the next engine in sequence returning to that engine after the sequence is finished.

5.3.2.1.2 Even Numbered Engines - The even numbered engines are to be started by windmilling. Beginning with the lowest numbered engine and proceeding in sequence, place the start switch in the "Ignite-Crank" position and hold. Move the throttle to the "Idle-Run" position. The helicopter rotor speed must be between 43 r.p.m. and 58 r.p.m. for windmill starting.

5.3.2.2 Operation - With all engines started and idling, move all the throttles to the "run" position. When the helicopter rotor speed approximates the minimum governed speed, move the master mode

selector switch to "select." The rotor speed should stabilize at the governed speed, and minor relative adjustments may be made with the throttles so that all engines are operating alike. Select the desired rotor speed with the rotor speed selector and allow the engines and rotor speed to stabilize.

5.3.2.2.1 Automatic Operation - In automatic mode, a movement of the collective stick, which causes the rotor blade pitch to increase, causes the collective power control to send a signal to the electronics package which, in turn, sends a signal to the automatic mode actuator causing it to actuate the fuel controls to increase power. As the rotor speed tends to deviate from the governed rotor speed, the electronics package sends additional signals as required to hold the rotor speed constant. A movement of the collective stick, which causes the blade pitch to decrease, causes the collective power control to send a signal to decrease power, and the electronics package monitors rotor speed and governed speed and sends signals as required to maintain rotor speed constant. Minor power adjustments may be made using the throttles so that all engines will approach their operating limits uniformly when maximum power is used.

5.3.2.2.2 Mixed Mode Operation - Mixed mode operation will generally result from a failure of a component such as an automatic actuator. The engine which fails to respond is placed in the manual mode with its mode selector. The only difference between this mode and automatic mode is that one engine is controlled manually. Signals from the collective power control are sent to the manual actuator for the engine in manual mode, and the manual trim is used to balance the power on the one engine with the engines in automatic mode. Rotor speed is controlled by the engines in automatic mode in the normal manner.

5.3.2.2.3 Manual Mode Operation - Operation in the manual mode will result from a failure such as the rotor speed sensor. Failure to hold rotor speed would be noted and the master mode selector would be placed in the "manual" position throwing all engines into the manual mode. A movement of the collective stick, which causes an increase in blade pitch, causes the collective power control to send signals to all the manual actuators to increase power, and a movement of the collective stick, which causes blade pitch to decrease, causes the collective power control to send signals to all the manual actuators to decrease power. The rotor speed is monitored by the pilot, and rotor speed is maintained to the desired value by means of the collective manual trim. Minor relative power adjustments may be made using the throttles so that all engines will approach their operating limits uniformly when maximum power is used.

5.3.2.3 **From Operation to Idle** - With the collective stick set for flat pitch, the rotor speed selector is set for lowest governed speed. In manual mode and mixed mode, the collective manual trim should be turned to its lowest power demand. In automatic and mixed mode, the master mode selector switch should be placed in the "manual" position. The throttles can be moved now to the "Idle-Start" position.

5.3.2.4 **Engine Shutdown** - With all engines idling, moving the throttle to the stop position causes a signal to the manual actuators to move them to the shutoff position.

5.3.3 **System Description** -

5.3.3.1 **Operating Modes** - The system shall have an automatic mode and a manual mode.

5.3.3.1.1 **Mode Selector** - A mode selector shall be provided which shall consist of a switch for each engine by which the engine can be placed in either mode. Each switch shall have two positions which shall be designated "manual" and "auto."

5.3.3.1.1.1 **Master Mode Selector Switch** - A master mode selector switch shall be provided which will simultaneously place all engines in either mode. It shall have two positions which shall be designated "Manual" and "Select."

5.3.3.1.1.2 **Switch Authority** - The master mode selector switch shall have overriding authority when in the manual mode position. When the master mode selector switch is in the select position, the individual switches shall have full authority.

5.3.3.1.2 **Manual Mode Operation** - With the master mode selector switch in either position and all the individual switches in the "manual" position, engine power shall be controlled manually by means of the collective power control and the collective manual trim.

5.3.3.1.3 **Automatic Mode Operation** - With the master mode selector switch in the "select" position and all the individual switches in the "automatic" position, engine power shall be controlled automatically so as to control the helicopter rotor speed to a pre-selected value.

5.3.3.1.4 **Mixed Mode Operation** - With the master mode selector switch in the "select" position and with some of the individual switches in the "automatic" position and some in the "manual" position, those engines whose mode selector switches are in the "manual" position shall have their power controlled manually; however, the engines whose mode selector switches are in the "automatic" position

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shall have their power controlled automatically so as to control the helicopter rotor speed to the pre-selected value.

5.3.3.2 Actuator - A dual actuator or two individual actuators shall be provided for each engine and shall mount on the engine fuel control. The actuator(s) shall actuate the engine fuel control and shall contain the necessary feedback device to enable it to function in the closed loop system. A brushless design is preferred.

5.3.3.2.1 Definition -

5.3.3.2.1.1 Manual Actuator - In this specification the term "Manual Actuator" shall refer to that portion of a dual actuator which shall be designated a manual actuator or to that actuator of two actuators which shall be designated a manual actuator.

5.3.3.2.1.2 Automatic Actuator - In this specification the term "Automatic Actuator" shall refer to that portion of a dual actuator which shall be designated an automatic actuator or to that actuator of two actuators which shall be designated an automatic actuator.

5.3.3.2.2 Operation -

5.3.3.2.2.1 Manual Actuator - The manual actuator shall be operative whenever that portion of the control system pertaining to the engine on which it is mounted is in the manual mode. It shall receive all signals, when operative, to increase or decrease power and shall actuate the fuel control accordingly. It shall be capable of actuating the fuel control throughout its entire range of operation independent of the position of the automatic actuator.

5.3.3.2.2.2 Automatic Actuator - The automatic actuator shall be operative when that portion of the control system pertaining to the engine on which it is mounted is in the automatic mode. It shall receive all signals, when operative, to increase or decrease power and shall actuate the fuel control accordingly. It shall be capable of actuating the fuel control throughout its entire range of operation independent of the position of the manual actuator in the normal operation range.

5.3.3.3 Collective Power Control - Major engine power change signals shall originate in the collective power control. The collective power control shall be actuated by the collective stick such that a movement of the collective stick which increases the helicopter rotor blade pitch will result in a signal to increase engine power, and a movement of the collective stick which decreases blade pitch will result in a signal to decrease engine power.

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5.3.3.3.1 **Manual Mode Operation** - During manual mode operation, signals originating in the collective power control shall be fed directly to the manual actuator(s) and shall cause a proportional actuation of the engine fuel control(s).

5.3.3.3.2 **Automatic Mode Operation** - During automatic mode operation, signals originating in the collective power control shall be fed to an electronics package which shall cause a signal to be sent to the automatic mode actuator which shall cause a proportional actuation of the engine fuel control.

5.3.3.3.3 **Mixed Mode Operation** -- During mixed mode operation, those engines in manual mode will be handled per 5.3.3.3.1 and those in automatic mode will be handled per 5.3.3.3.2.

5.3.3.4 **Collective Manual Trim** - The collective manual trim shall be operative on all engines during manual mode operation, and during mixed operation it shall be operative on those engines which ar in manual mode. Manual trim shall be accomplished by means of a twist-grip handle mounted on the collective stick. It shall cause a proportional movement of the manual actuators, which are operative, by means of a signal sent directly to the manual actuators or by modifying the signals from the collective power control. Configuration of the grip and direction of operation are subject to Hiller approval.

5.3.3.5 **Rotor Speed Selector** - The rotor speed selector shall be capable of varying rotor-governed speed during automatic and mixed mode operation from 50-percent to 110-percent rated rotor speed. Configuration of the operating control shall be subject to Hiller approval.

5.3.3.6 **Rotor Speed Sensor** - The rotor speed sensor shall sense rotor speed during automatic and mixed mode operation and send signals to the electronics package so that power will be increased or decreased based on rotor underspeed or overspeed, respectively, from the pre-selected value of rotor-governed speed.

5.3.3.7 **Throttles** - A throttle lever shall be provided for each engine, and each lever shall have three positions; namely, stop, idle-start, and run. The stop and idle-start positions shall be operative only on those engines which are in the manual mode.

5.3.3.7.1 **Throttle Stop Position** - Moving the throttle lever into the stop position shall cause the manual actuator to go beyond the idle-start position into the shutoff position. A detent shall be provided so that the throttle lever cannot accidently be moved to the stop position.

5.3.3.7.2 Throttle Idle-Start Position - Moving the throttle lever into the idle-start position causes the manual actuator to actuate the engine fuel control to the extreme low end of the normal operation range.

5.3.3.7.3 Throttle Run Position - In the run position, the throttles shall be used to vary the relative power between engines during manual and automatic operation and for full control of engine power during static ground running. In the manual mode, signals originating in the throttles shall be sent directly to the manual actuator or shall modify the signal from the collective power control. In the automatic mode, signals originating in the throttles shall be sent to the electronics package.

5.3.3.8 Electronics Package - The electronics package shall be designed per MIL-E-5400, Class 1A, Category II, and shall receive all automatic mode signals input to the system and combine them into a single signal for each engine to increase or decrease power on those engines which are in automatic mode. These signals shall be output to the respective fuel control automatic mode actuator.

5.3.3.9 Cyclic Power Control System - Consideration shall be given to a cyclic power control system to smooth out the cyclic variations of engine power and rotor draft which occur when the helicopter is in forward flight. This system shall not be incorporated in the power management system initially, but provisions shall be made for its incorporation in the future. Coordination with the engine fuel control and engine manufacturer will be required to insure compatibility; and all design, performance, and hardware shall be subject to Hiller approval.

5.3.3.10 Warning System - Provisions shall be made to supply 115-volt AC signals to operate a Hiller furnished warning device when the rotor speed deviates from the selected governed speed by as much as 5 percent. Separate signals shall be provided for overspeed and underspeed.

5.4 Operating Limits and Environment -

5.4.1 Operating Limits -

5.4.1.1 Rotor Speed - During automatic mode operation, the rotor speed shall be held within 3 percent of the selected rotor speed except at the power extremes where maximum power is inadequate or minimum power is excessive.

5.4.2 Environment -

5.4.2.1 Minimum Temperature - The minimum temperature shall be -65°F.

5.4.2.2 **Maximum Temperature** -

5.4.2.2.1 **Airframe-Mounted Components** - The maximum temperature of airframe-mounted components shall be $125^{\circ}F$.

5.4.2.2.2 **Rotor Tip-Mounted Components** - The maximum temperature of rotor-mounted components shall be $250^{\circ}F$.

5.4.2.3 **Pressure Altitude** - Components shall perform all required functions at all pressure altitudes between sea level and 20,000 feet.

5.4.2.4 **Acceleration Field** - Rotor tip-mounted components shall perform all required functions in a continuous acceleration field environment 235 times the acceleration of gravity. In addition, they shall be capable of performing satisfactorily in a continuous acceleration field environment 259 times the acceleration of gravity for 30 seconds with a cumulative total time of 30 minutes per 1,000 hours of operation. Structural integrity shall be maintained under a continuous acceleration field environment 367 times the acceleration of gravity.

5.5 **General Requirements** -

5.5.1 **Electrical Requirements** -

5.5.1.1 **Compatibility** - All electrical components shall be compatible with a 115-volt, AC, 400-cycle power supply per MIL-STD-104 and shall be considered to be category "B" utilization equipment.

5.5.1.2 **Modules** - Wherever possible, components shall be designed with conveniently replaceable, identical modules for the portions of the circuits which perform like functions.

5.5.1.3 **Connectors** - All electrical connectors shall be of the bayonet coupling type per MIL-C-26482.

5.5.2 **Mechanical** -

5.5.2.1 **Compatibility** - The shapes, colors, and materials of all knobs and grips shall be subject to Hiller approval.

5.5.3 **Dynamics** - The system response shall be compatible with the engine, the engine fuel control, the main rotor assembly, the tail rotor including the tail rotor power source and the airframe such that the resulting system is stable.

5.5.4 **Reliability and Fail Safety** -

5.5.4.1 Reliability - Reliability shall be considered of major impor-
tance and of equal importance with fail safety. All systems and
components procured under this specification shall have a minimum
service life of 1,200 flight hours. Vendors shall guarantee flight
prototypes to be maintenance free for a minimum of 50 flight hours.

5.5.4.2 Fail Safety - Fail safety shall be subject to Hiller approval.

5.5.4.2.1 Electrical - Components and internal circuitry shall be
designated so that loss of one circuit shall not result in the
loss of another circuit, the system, or the aircraft, nor diminish
aircraft controllability.

5.5.4.2.2 Mechanical - Failure of any mechanical component shall not
result in the failure of any other mechanical component. It shall
not result in loss of ability to control any engine in its normal
range of operation except that the failure of any geared device
common to a manual actuator and to an automatic actuator used to
actuate an engine fuel control may result in loss of ability to
control the engine on which it is installed. No single mechanical
failure shall diminish aircraft controllability.

All mechanical components experiencing stress or motion as a func-
tion of its use, whose failure would result in loss of another
component, loss of control of an engine or diminish aircraft con-
trollability shall be designed with a minimum safety factor of 2.5
based on the following: material yield strength, material fatigue
endurance limit or 10^6 cycle fatigue strength (as applicable),
1,000-hour stress rupture properties, 2-percent elongation in 1,000
hours' creep strength, bearing strength, seizure loading for slid-
ing parts, and B-10 service life of 1,200 hours for antifriction
bearings as critical and as applicable for the worst condition of
loading that can be experienced during periods of operation. Any
deviation from the above shall be subject to Hiller approval.

5.5.5 Development Prototypes - With prior Hiller approval, development
prototypes of electronics packages may be constructed in "bread
board" configurations.

5.5.6 Other Requirements - Other requirements such as data and draw-
ings per MIL-D-70327, Environment Testing per MIL-E-5272 or MIL-T-
5422, Identification per MIL-STD-130, Interchangeability per MIL-
I-8500, Interference Limits per MIL-I-26600 or MIL-I-6181, Elec-
tronic Equipment Design per MIL-E-5400, Test Reports per MIL-T-9107,
Quality Control System requirements per MIL-Q-9858, and Reliability
per MIL-R-27542, MIL-HDBK-217, MIL-STD-756 and MIL-STD 721 will
apply in that all units shall be designed so that they could be
qualified per these specifications.

6.0 APPENDIX II

HILLER MODEL 1108 HELICOPTER OIL COOLER DESIGN SUMMARY

Cooling Air Conditions

Maximum available air pressure at maximum cooling condition	6.0 in. H_2O
Ambient temperature	$115^{\circ}F$.
Recirculation temperature rise	$10^{\circ}F$.
Ram temperature rise	$60^{\circ}F$.
Oil cooler air inlet temperature	$185^{\circ}F$.

Oil Conditions

Engine oil inlet temperature (max.) . . .	$225^{\circ}F$.
Oil flow rate	5.5 g.p.m.
Engine heat rejection to oil	1,200 BTU/min.
Oil type	MIL-L-7808

Oil Cooler Recommendation by Harrison Division, GMC

Core size:

Cooler depth	3.75 in.
Cooler length	13.0 in.
Cooler height	8.0 in. (16 in. for a two-core cooler)
Core weight	8.1 lb., dry

2-pass, cross-counter oil flow pattern

Required air flow rate	125 lb/min.
Oil pressure drop	15 p.s.i.

7.0 APPENDIX III

J69-T-29 TIP TURBOJ AFT SECTION COOLING

7.1 Introduction

The primary purpose of this study was to analyze the aft section cooling
requirements of the J69-T-29 tip turbojet engine installation, and to
provide a logical basis for the design of the cooling system if its use
was found necessary. Suggestions are offered for forward section venti-
lation. The prime cooling system components that were studied were the
cooling air inlet and the means of pumping the cooling flow. The exhaust-
powered ejector pump and ram inlet were investigated as two means of
overcoming the system pressure loss. Other possible systems that were
ruled out after cursory examination were compressor bleed, auxiliary
blower, and centrifugal pumping.

Section 7.2 discusses the cooling requirement while Section 7.3 dis-
cusses the means of pumping cooling flow. Section 7.4 is a discussion
of the forward section ventilation.

7.2 Cooling Analysis

The prime results of the cooling analysis are presented in Tables 1 and
2 and in Figure 33. The more controversial assumptions are included in
the tabulations.

In the case of the aluminum shroud-nacelle combination, the shroud-
nacelle is assumed to be at a constant 250°F. throughout. In the case
of the titanium shroud, the nacelle is assumed to have a maximum temper-
ature of 800°F. at the exit. In the heat transfer calculations, the
engine, itself, is assumed to be an infinite heat source; i.e., the
engine shell temperatures are assumed constant, independent of the heat
flow (or rejection) from the engine. The engine temperatures used were
those supplied as representative of the J69-T-29 engine.

Table 1 details the heat rejection by the engine to the shroud or nacelle
depending on the configuration. In Table 1 the assumption requiring the
closest scrutiny is the emissivity assumed for the aluminum shroud-na-
celle combination. Highly polished aluminum plate is known to have
emissivity in the order of .045 at 250°F. while rough plate has an
emissivity of approximately .06 at this temperature. Aluminum that has
been oxidized at 1,100°F. is known to have an emissivity of .11 at rough-
ly 250°F. The detail effects of weathering and service on the emissiv-
ity of aluminum alloy are not known at this time. Effort is being made
to obtain such information.

Table 2 details the heat loss by the nacelle to its ambient surroundings.
There are no critical emissivity assumptions in this case, in that any

TABLE 1
ENGINE HEAT REJECTION

250°F. Alum. Shroud-Nacelle

Shroud emissivity $= .11$
Engine emissivity $= .43$ at 725°F.; $= .75$ at 1,300°F.

$q_{radiation}$	$= 7,000$ BTU/hr-engine
$q_{conduction}$ through stagnant air between engine and shroud	$= 1,360$ BTU/hr-engine
q_{total}	$= 8,360$ BTU/hr-engine

3/4-Inch Blanket on Engine

$k = \dfrac{.58 \text{ BTU-in.}}{\text{hr-ft}^2\text{-}°F.}$, Ref: H.I. Thompson's F.A. Batt insulation, 6 lb/ft^3

$c = .21$ BTU/lb-°F.

$q_{conduction} = 287 \dfrac{\text{BTU}}{\text{hr-ft}^2_{nacelle}}$ ave. over engine surface

$= 680 \dfrac{\text{BTU}}{\text{hr-ft}^2_{nacelle}}$ over tail cone

800°F. Max., Titanium Nacelle

Nacelle emissivity $= .7$
Engine emissivity $= .43$ at 725°F.; $= .75$ at 1,300°F.

$q_{radiation}$	$= 12,800$ BTU/hr-engine
$q_{conduction}$ through stagnant air between engine and shroud	$= 1,000$ BTU/hr-engine
q_{total}	$= 13,800$ BTU/hr-engine

TABLE 2
NACELLE HEAT LOSS

$T_{amb} = 117°F$, full sun 40° lat., zero ambient wind, nacelle closed

Convection + Radiation - Solar Heating = Total

Condition	Tip Speed (f.p.s.)	Convection (BTU/hr.)	Radiation (BTU/hr.)	Solar Heating (BTU/hr.)	Total $\left(\dfrac{BTU/hr.}{Engine}\right)$ or $\left(\dfrac{BTU/hr.}{ft^2 \text{nacelle}}\right)$	
250°F. Skin Temp. *	Static	1,530 (h = .9)	2,920 (ε = .91)	310 (ε = .16)	4,140	324
	20 (induced nacelle surface velocity)	4,050 (h = 7.38)	2,920	310	6,660	520
	204 (30 rotor r.p.m.)	31,500 (h = 18.5)	2,920	310	34,110	2,660
450°F. Skin Temp.	Static	4,900 (h = 1.15)	11,300 (ε = .88)	1,900 (ε = 1)	14,300	1,120
800°F. MAXIMUM Skin Temp. **	Static	8,500 (h = 1.22 to 1.35)	12,300 (ε = .7)	1,900 (ε = 1)	18,900	1,480
	204 (30 rotor r.p.m.)	129,000 (h = 18.5)	12,300	1,900	139,400	10,890

* Aluminum construction (painted white)
** Titanium construction

$T_{amb.} = 117^\circ F.$, military power, $\dot{w}_{cooling} = 1$ percent $\dot{w}_{primary}$, aluminum construction.

Engine - $1300^\circ F.$
Shroud - $250^\circ F.$
Nacelle - $225^\circ F.$

Stagnant air
Cooling air
.093-inch web
.84 inch
16 channels
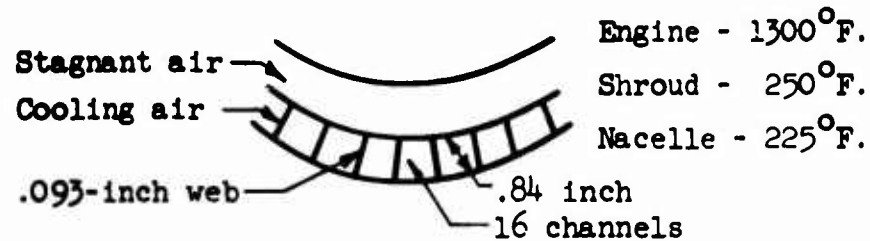
$V_{cooling} = 14.5$ f.p.s.

$T_{cooling\ air\ rise} = 25^\circ F.$

$q_{total} = 15,000$ BTU/hr-engine

Engine - $1300^\circ F.$
Shroud - $250^\circ F.$
Nacelle - $230^\circ F.$

Cooling air

Convective heat transfer from engine to cooling air is high, due to large temperature drop $(1,300^\circ F. - 250^\circ F.)$ across surface film. At 6-percent cooling flow, cooling air temperature rise is such as to preclude adequate cooling of shroud which is subject to radiant heating by engine.

Figure 33. Auxiliary Nacelle Cooling.

increase in emissivity increases the amount of heat loss from the na-
celle which, in turn, tends to reduce the nacelle temperature. The
convective heat transfer coefficients for forced convection were checked
and found to be compatible with those of Reference 22. The values for
the nacelle heat loss take into account the fact that, in a dual nacelle,
100 percent of the nacelle surface is not effective in providing an
area for heat loss. In this particular installation, the effective
area per engine has been computed to be approximately 87 percent of the
single-engine, basic nacelle. Also, only that nacelle area to the rear
of the firewall is considered, the forward section cooling being ana-
lyzed separately.

It will be noted from Table 2 that the engine starting technique has
tremendous influence on the nacelle heat loss. The static condition
is equivalent to the initial nacelle heat loss in the event of an air
impingement start of the turbojet. The 205-foot-per-second (30 r.p.m.)
case is indicative of the nacelle heat loss in the event of a windmill
start of the turbojet. The 20-foot-per-second case is an estimation of
the nacelle heat loss due to the flow over the nacelle produced by free
jet pumping.

Figure 33 details the cooling characteristics of two auxiliary cooling
systems. In the first, the cooling air is passed through cooling pas-
sages between a radiant or reflecting shroud and the nacelle skin; and
in the second system, the cooling air is passed between the shroud and
the engine. In the analysis of the first system, it was assumed that
the cooling passages run fore and aft on the engine. Such construction,
while not the most satisfactory structurally, will provide better cool-
ing than hoopwise-oriented cooling passages. The second system is im-
practical because it tries, in effect, to cool the engine.

Combining Tables 1 and 2 and Figure 33 indicates several solutions to
the problem of maintaining desired nacelle temperatures. For instance,
from Table 1 it is seen that in order to maintain a 250°F. nacelle tem-
perature without benefit of insulation, 8,360 BTU/hour per engine will
have to be removed from the nacelle. Table 2 shows, in the case of an
air impingement start (static), that the heat rejection of the nacelle
to ambient is in the order of half that required to maintain the nacelle
at 250°F. A minimum of auxiliary cooling, as shown in Figure 33, in
addition to the static heat loss of the nacelle, results in an an ade-
quate margin over the heat rejection required to maintain the 250°F.
nacelle temperature. Tables 1 and 2 also show that the air impingement
start can be tolerated if an insulating blanket in the order of 3/4
inches thick is applied to the engine aft section. The thermal lag of
the insulation and nacelle will prevent temperature excursions above
the 250°F. level in the critical area of the tail cone until forced
convection (windmill) cooling is adequate to supply the cooling re-
quired. The overall lag between light-off and 250°F. nacelle tempera-

ture is estimated to be in excess of 120 seconds assuming an effective
nacelle mass thickness of .060 inch. Whether or not an insulating
blanket is used, a minimal ventilating air flow should be provided.
It is possible that the thermal lag of the engine shell in combination
with the thermal lag of the nacelle is adequate to limit the nacelle
temperature to 250°F. until forced convection (or windmill) cooling is
adequate. It should be noted from Tables 1 and 2 that there is no prob-
lem in maintaining a 250°F. nacelle temperature without the benefit of
insulation or auxiliary cooling if the windmill start of the turbojet
is employed.

If a titanium nacelle is assumed with 800°F. maximum temperature,
Tables 1 and 2 show that the static heat loss of the nacelle has a
reasonable margin over the engine heat rejection to the nacelle. How-
ever, use of the titanium nacelle will require titanium or stainless
steel blade tip construction, due to proximity of hot nacelle parts.
Also, some provision may be required to cool the "shaded" nacelle por-
tions because of the low conductivity of titanium.

7.3 Means of Pumping Cooling Flow

Analysis of cooling requirements of the J69-T-29 tip turbojet installa-
tion in Section 7.2 has shown that a cooling flow in the order of one
to two percent of the primary flow rate of the engine is more than ade-
quate to provide the cooling required. In the following discussion both
ram inlets and ejector pumping systems are considered.

7.3.1 Ram Inlet

The use of a ram inlet to provide the pumping pressure rise required for
a cooling flow rate of two percent of the engine flow rate would cause a
thrust penalty in the order of one percent of the engine net thrust at a
rotor tip speed of 700 feet per second. The one-percent thrust loss as-
sumes that all the inlet momentum of the cooling air is dissipated in
overcoming flow losses through the cooling system; i.e., cooling air
inlet velocity equals V_o, and the cooling air outlet velocity equals
zero. The ram inlet system is obviously inadequate for static cooling.

7.3.2 Ejector Pump System

Applicable NASA cooling ejector data (References 23 through 31) were
reviewed and analyzed. The data for a series of conical shroud ejector
configurations (References 23 and 24), which are presented in Figures 34
through 41, were corrected for inlet momentum considerations and plotted
as a function of weight flow ratio and secondary stream total pressure
at the ejector face. Figures 34 through 37 present the data for condi-
tions of military power at a forward speed of 700 feet per second. How-
ever, the data does not include the effect of air flow over tail section

on ejector characteristics, the effect of secondary base annulus static pressure resulting from forward flight on the net thrust, nor the effect of submerged boundary layer inlets on nacelle drag reduction. Figures 38 through 41 present data for conditions of static operation and military power. The method of data correction is presented in Section 7.3.3.

These data, with the exception of those for a diameter ratio of 1.4, support the statement on page 332 of Reference 32; namely, that the diameter ratio (shroud to turbojet exit) and shroud overhang (expressed as a ratio to the turbojet exit diameter) should be held to a minimum commensurate with the cooling flow requirements to achieve maximum net thrust performance. (The behavior of the data for the 1.4 diameter ratio configuration is unexplained.)

References 33 through 39 indicate that the effect of air flow over the tail section on ejector performance was negligible. However, the data documenting the effect of forward flight produced base pressure with ejector configurations was less conclusive. A varying base force of approximately ±2 percent of the total gross still air ejector force was quoted by NASA investigators (Reference 33). Such a value is probably not within the overall accuracy of the ejector force data. It is believed that this base pressure effect can be considered negligible, particularly for an ejector configuration incorporating minimum (i.e., optimum) diameter ratios. However, in light of the NASA results, the base pressure should be checked in wind-tunnel tests of the prototype nacelle configuration.

References 28, 30, and 40 describe the effect of elevated primary jet temperatures on cooling ejector performance. In general, the thrust performance was not affected, but corrected weight flow ratios should be obtained from the weight flow parameter,

$$\frac{\dot{w}_s}{\overline{w}_p} \sqrt{\frac{T_s}{T_p}}$$

Figure 34 shows the improvement in thrust performance that can be realized from lower total pressures at the ejector face. This leads to the conclusion that use of submerged (or boundary layer) inlets for the cooling air not only improves the cooling system performance, but also improves nacelle performance through drag reduction by decreasing the boundary layer thickness. References 41, 42, and 43 discuss boundary layer inlets and their matching to ejector requirements.

In regard to the design of the tip turbojet nacelle exit, the following comments are offered:

a) The existing 1.5 diameter ratio is obviously far in excess of any requirement of an ejector pumping system for the cooling requirements of a non-afterburning turbojet.

b) The performance of such a configuration is difficult to pinpoint in that it is far from any configuration tested at the cooling flow rates of interest to this installation (losses as high as 10% are indicated).

c) The exit diameter should be reduced as much as possible commensurate with external drag considerations. A diameter ratio of 1.1 would be most satisfactory from the ejector standpoint.

d) The data indicates that the shroud overhang should be between 40 percent and 80 percent of the turbojet exit diameter.

7.3.3 Ejector Data Presentation

NACA ejector data is presented in terms of

$$\frac{F_{ej}}{F_j}, \quad \frac{\dot{w}_s}{\dot{w}_p}\sqrt{\frac{T_s}{T_p}}, \quad \frac{P_s}{P_o} \quad \text{and} \quad \frac{P_p}{P_o}.$$

The ratio of the gross ejector thrust (F_{ej}) to the gross thrust of primary nozzle without shroud (F_j) does not properly account for the influence of inlet momentum on ejector performance.

Therefore, the following correction was applied to the NACA data. To properly assess ejector performance, the ratio of total net ejector thrust $(F_{N_{ej}})$ to the net thrust of the primary nozzle (the turbojet net thrust) without shroud (F_{N_j}) is required. This ratio is obtained from the following expression:

$$\frac{F_{N_{ej}}}{F_{N_j}} = \frac{F_{ej} - \left[\left(\begin{array}{c}\text{Turbojet inlet}\\\text{momentum}\end{array}\right)(D_{r_j}) + \left(\begin{array}{c}\text{Secondary flow}\\\text{inlet momentum}\end{array}\right)(D_{r_s})\right]}{F_j - \left[\text{Turbojet inlet momentum } (D_{r_j})\right]}$$

or

$$\frac{F_{N_{ej}}}{F_{N_j}} = \frac{\dfrac{F_{ej}}{F_j} - \left[\dfrac{D_{r_j}}{F_j} + \dfrac{D_{r_s}}{F_j}\right]}{1 - \dfrac{D_{r_j}}{F_j}}$$

where

$$\frac{D_{r_j}}{F_j} = \frac{\dot{w}_p V_o}{\dot{w}_p V_j} = .322 \qquad \text{at Mil. Power}$$

$$V_o = 700 \text{ f.p.s.}$$
$$\dot{w}_p = 34 \text{ lb/sec.}$$
$$F_{N_j} = 1,560 \text{ lb.}$$

Also

$$\frac{D_{r_s}}{F_j} = \frac{\dot{w}_p V_o}{\dot{w}_p V_j}$$

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where

$$V_o = \sqrt{2g\ Jc_p}\sqrt{T_o}\ \sqrt{1 - \left(\frac{P_o}{P_o}\right)^{\frac{\gamma-1}{\gamma}}}$$

$$V_j = \sqrt{2g\ Jc_p}\sqrt{T_p}\ \sqrt{1 - \left(\frac{P_o}{P_o}\right)^{\frac{\gamma-1}{\gamma}}}$$

Assuming $T_o = T_s$, and that the gas properties are equal for both streams, dividing V_o by V_j gives

$$\frac{V_o}{V_j} = \sqrt{\frac{T_s}{T_p}}\ \sqrt{\frac{1 - \left(\frac{P_o}{P_o}\right)^{\frac{\gamma-1}{\gamma}}}{1 - \left(\frac{P_o}{P_p}\right)^{\frac{\gamma-1}{\gamma}}}}$$

Combining gives

$$\frac{D_{r_s}}{F_j} = \left[\frac{\dot{w}_s}{\dot{w}_p}\sqrt{\frac{T_s}{T_p}}\right]\ \sqrt{\frac{1 - \left(\frac{P_o}{P_o}\right)^{\frac{\gamma-1}{\gamma}}}{1 - \left(\frac{P_o}{P_p}\right)^{\frac{\gamma-1}{\gamma}}}}$$

Finally, for a $V_o = 700$ f.p.s.,

$$\frac{F_{N_{ej}}}{F_{N_j}} = \frac{\dfrac{F_{ej}}{F_j} - \left[.322 + \dfrac{\dot{w}_s}{\dot{w}_p}\sqrt{\dfrac{T_s}{T_p}}\ \sqrt{\dfrac{1 - \left(\frac{P_o}{P_o}\right)^{\frac{\gamma-1}{\gamma}}}{1 - \left(\frac{P_o}{P_p}\right)^{\frac{\gamma-1}{\gamma}}}}\right]}{(1 - .322)}$$

The efficiency of the secondary air duct system can be measured by P_s/P_o. The system efficiency may be introduced by

$$\frac{P_s}{P_o} = \eta$$

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or

$$\eta = \frac{P_o}{P_o} \frac{P_s}{P_o}$$

Since the NACA data was presented as a function of P_s/p_o, and the corrected data herein is for a constant value of P_o/p_o (equivalent to a tip speed of 700 feet per second), each value of P_s/p_o implies a specific value of the duct system efficiency η. Namely:

700 f.p.s. tip speed		Static operation	
$\dfrac{P_s}{P_o}$	η	$\dfrac{P_s}{P_o}$	η
		.9	.9
		.95	.95
		1.0	1.0
1.0	.766		
1.1	.843		
1.2	.918		

7.4 Forward Section Ventilation

It is suggested that the forward section be ventilated by main inlet flow through a series of orifices in the inlet duct just forward of the engine inlet flange. This ventilating air would be exhausted through a similar series of orifices in the nacelle skin just forward of the fire wall.

The combination of ram pressure rise and low surface pressures at the exit orifices will supply adequate pumping in forward flight. In static operation, the reduction of pressure in the inlet duct will draw air in the opposite direction to provide adequate ventilation. The maximum ventilating requirement occurs at maximum V_o where maximum pumping pressure will occur. The zero ventilating flow region will occur at a very low value of V_o and, consequently, should cause no problem.
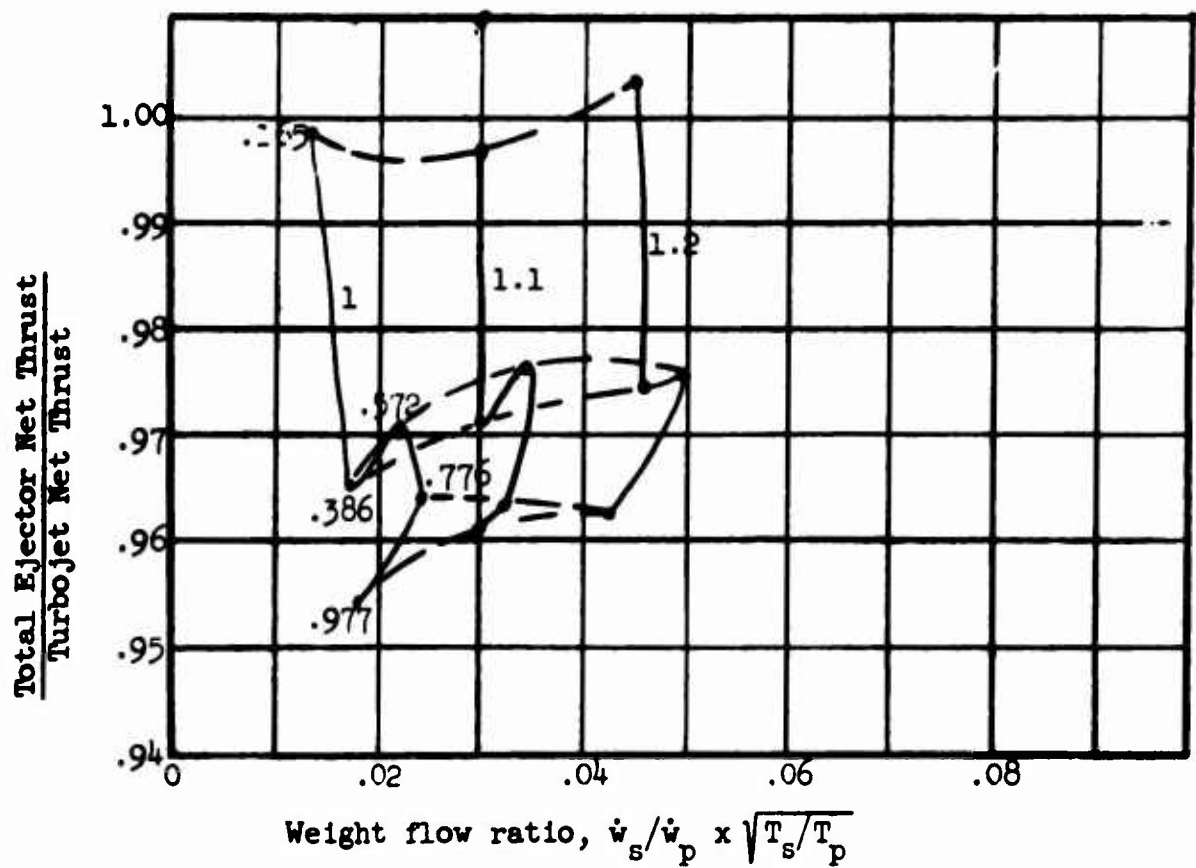
112
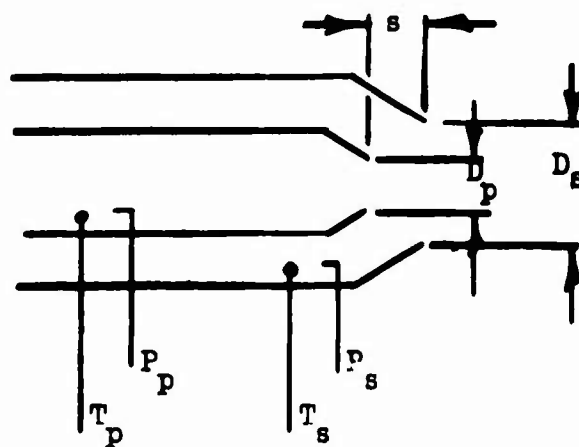
Figure 34. Thrust Performance - Conical Cooling Ejector.
$D_s/D_p = 1.06$

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Figure 35. Thrust Performance - Conical Cooling Ejector.
$D_s/D_p = 1.1$

Figure 36. Thrust Performance- Conical Cooling Ejector.
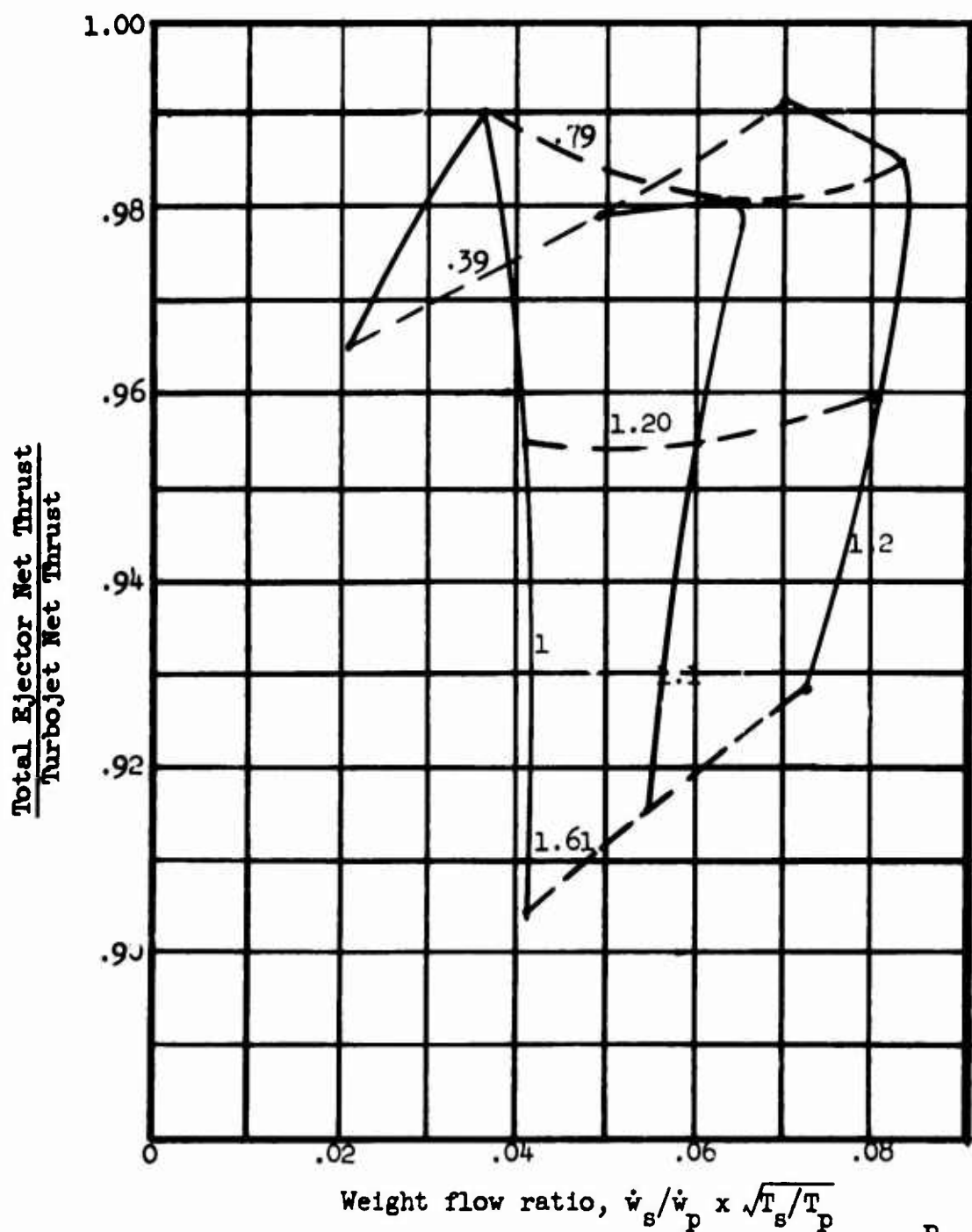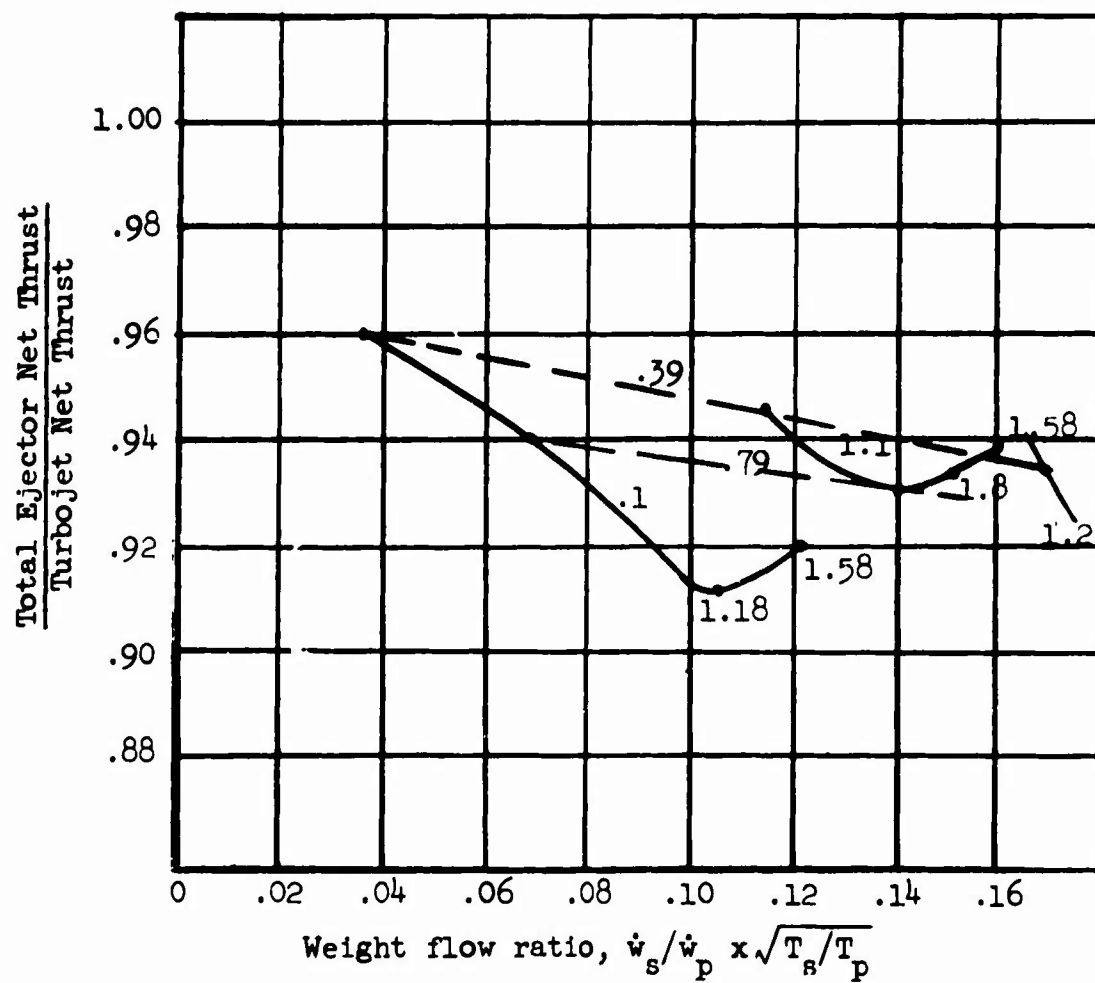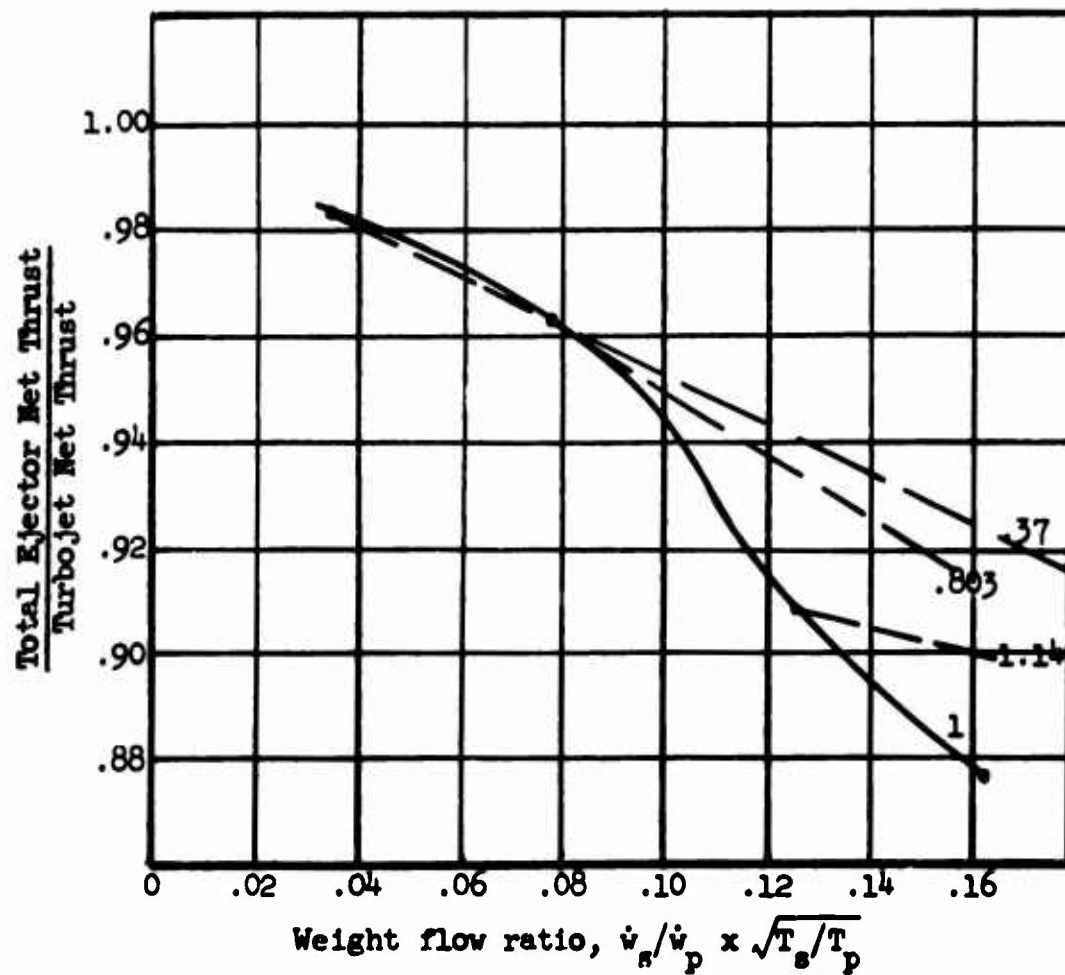$D_s/D_p = 1.21$

Figure 37. Thrust Performance - Conical Cooling Ejector.
$D_s/D_p = 1.4$

Solid lines —— indicate ejector pressure ratio, P_s/P_o

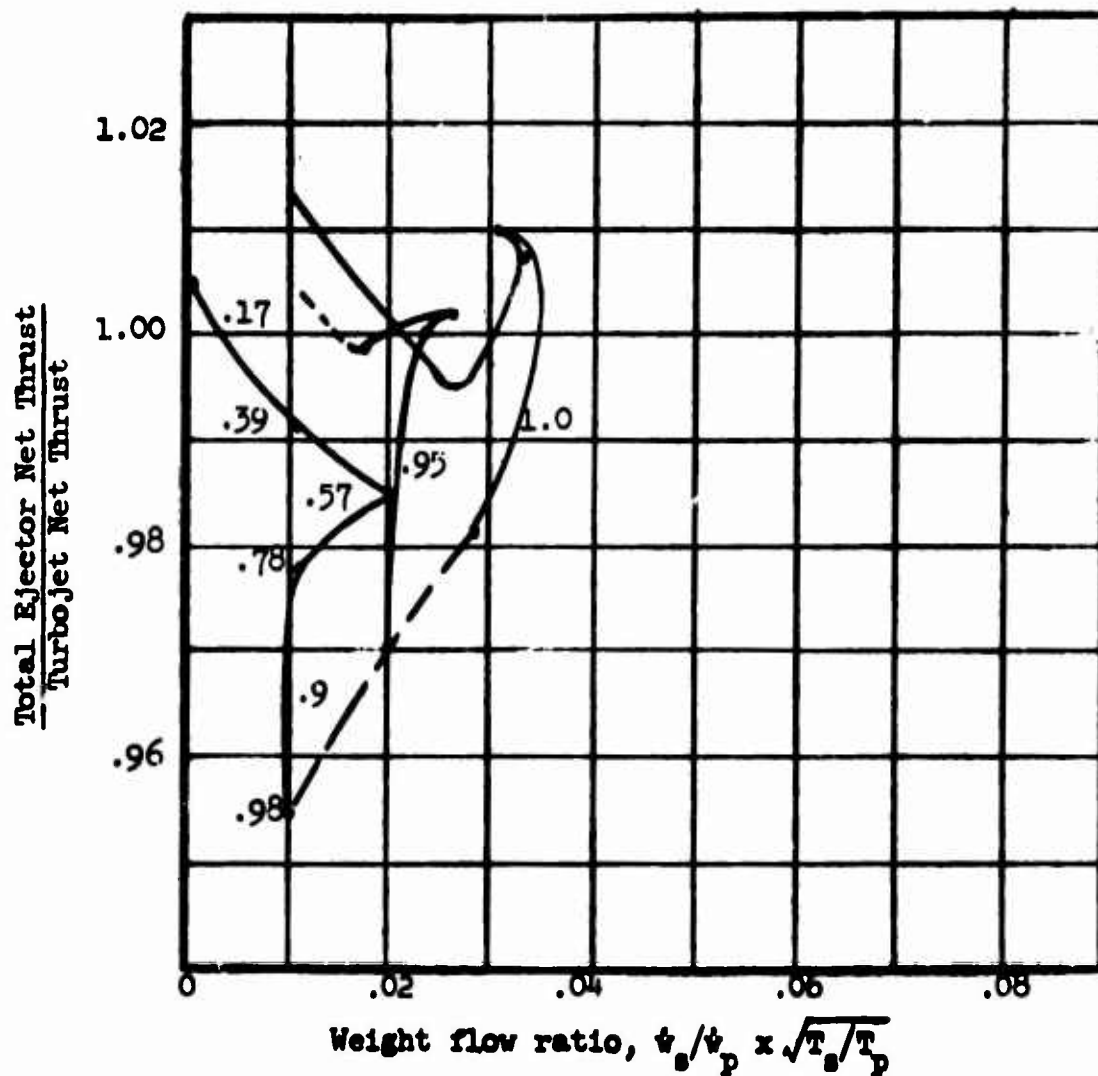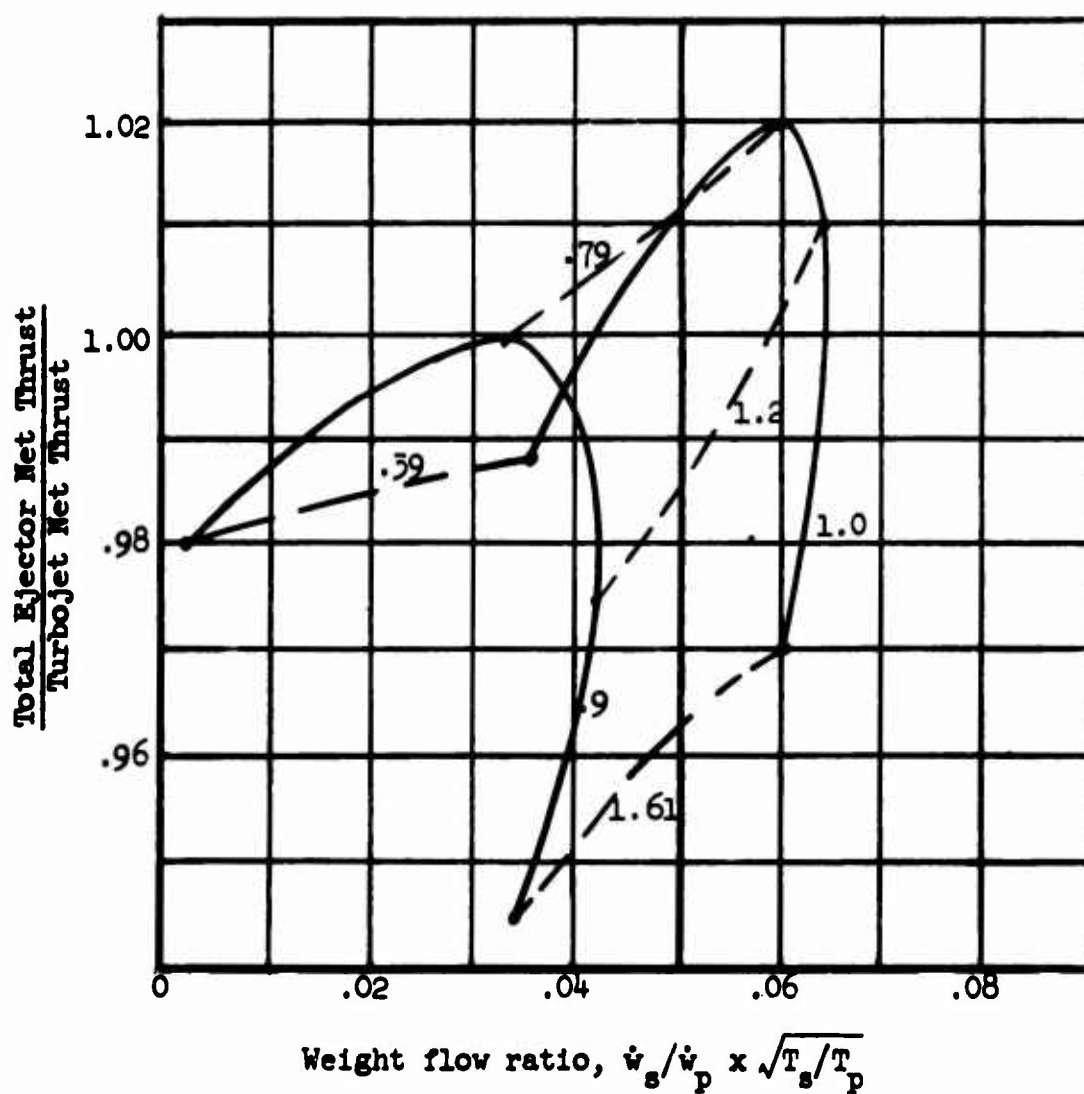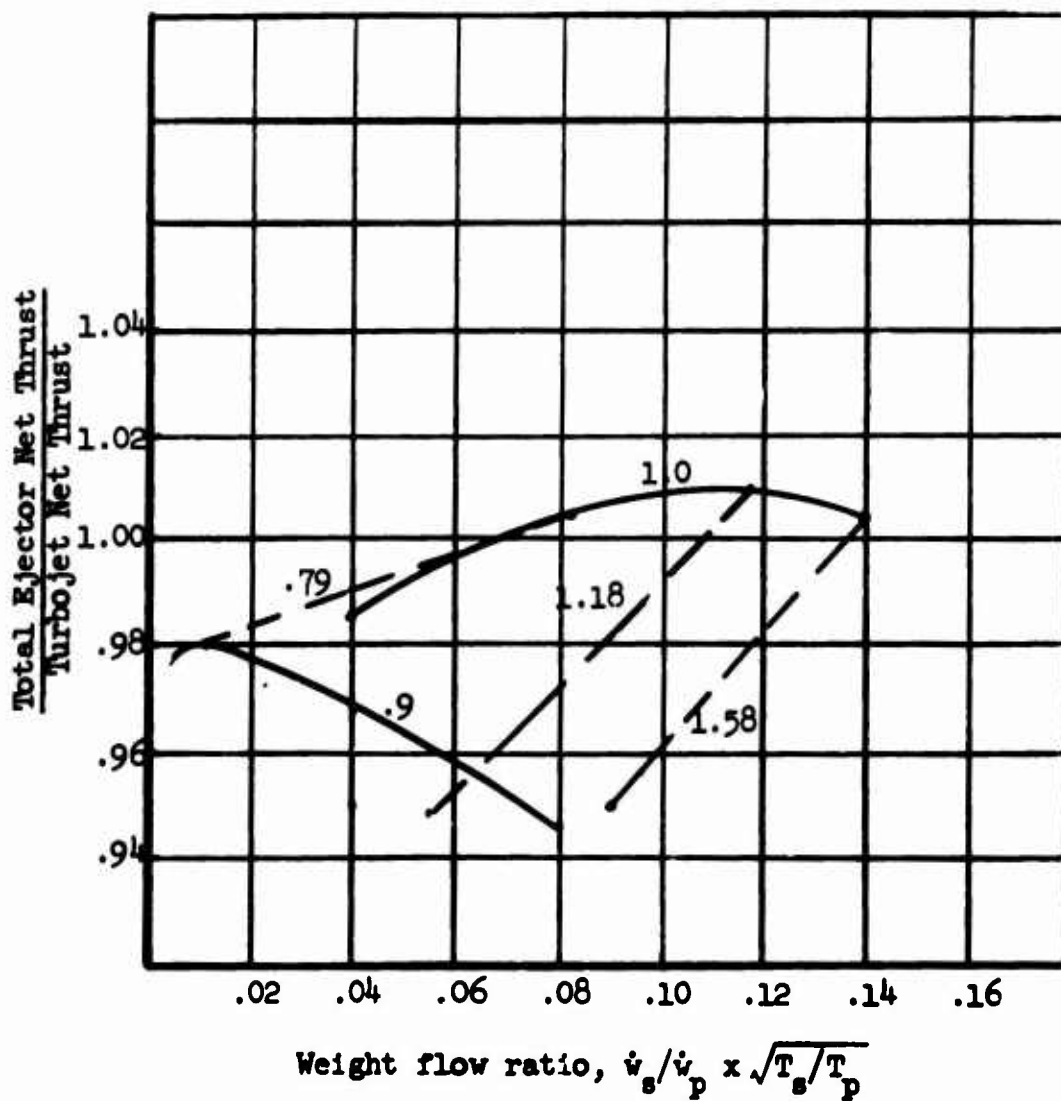Broken lines --- indicate spacing ratio, s/D_p

S.L. Std. - Mil. Power, $\dfrac{P_p}{P_o} = 2.0$
Static Operation

Figure 38. Thrust Performance - Conical Cooling Ejector.
$D_s/D_p = 1.06$

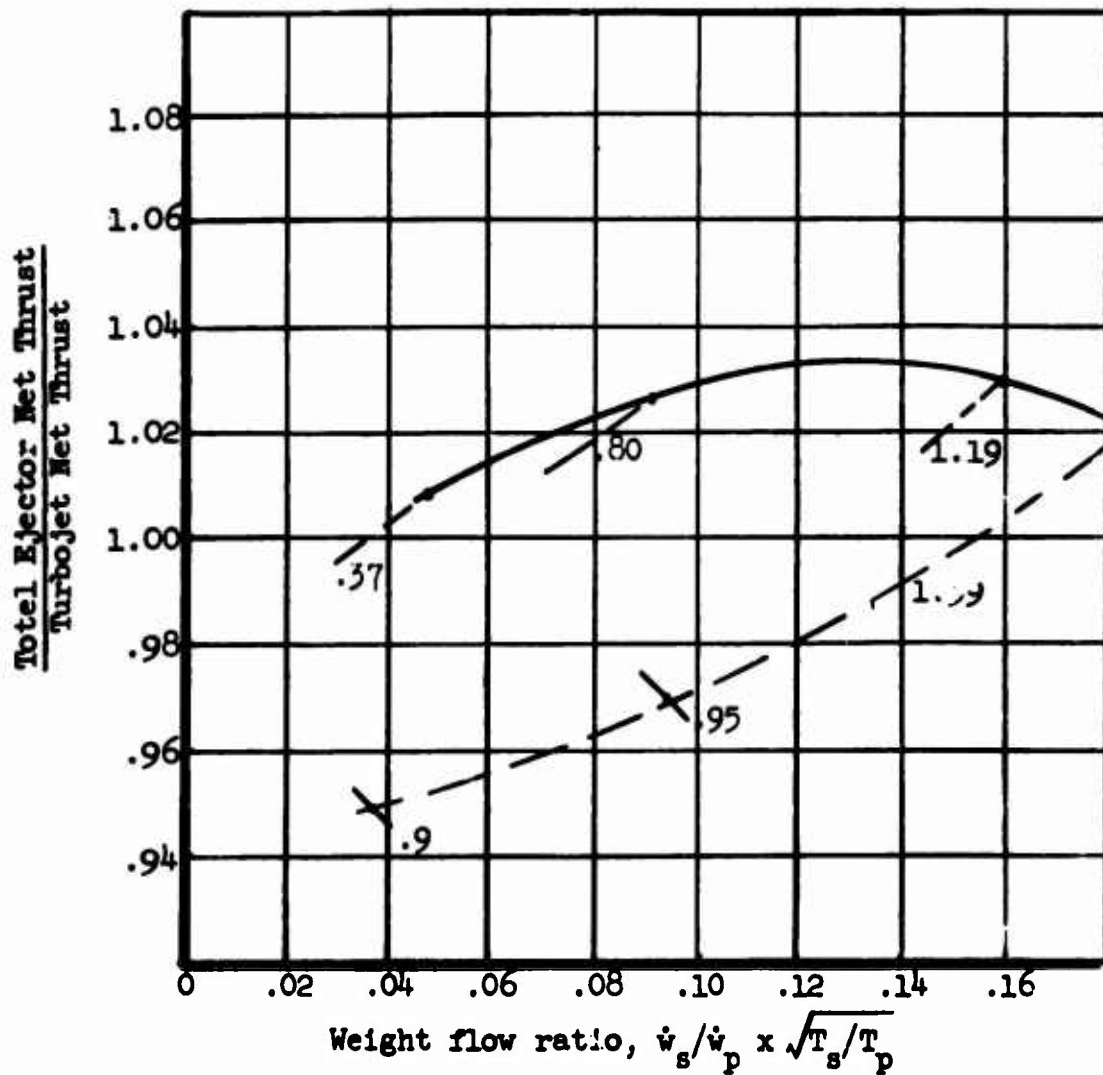Figure 39. Thrust Performance - Conical Cooling Ejector.
$$D_s/D_p = 1$$

Solid lines —— indicate ejector pressure ratio, P_s/P_o
Broken lines --- indicate spacing ratio, s/D_p

S.L. Std. - Mil. Power, $\dfrac{P_p}{P_o}$ = 2.0
Static Operation

Weight flow ratio, $\dot{w}_s/\dot{w}_p \times \sqrt{T_s/T_p}$

Total Ejector Net Thrust / Turbojet Net Thrust

Figure 40. Thrust Performance - Conical Cooling Ejector.
$D_s/D_p = 1.21$

Figure 41. Thrust Performance - Conical Cooling Ejector.
$D_s/D_p = 1.4$

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Volume III discusses the design studies of the heavy-lift tip turbojet rotor system for a heavy-lift crane-type helicopter. Design investigations were made primarily of rotor hub and blade retention, flight controls, rotor blade structure, power plant installation, and required systems such as fuel, etc.

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